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SESSION P

Electric Propulsion for Spacecraft

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WOLFGANG E. MOECKEL, *Chief of the Electromagnetic Propulsion Division of the NASA Lewis Research Center, is currently concerned with most of NASA's research on electric propulsion systems suitable for advanced space missions, as well as studies in plasma physics, magnetics, and space-environment simulation. Among his contributions to research are the use of aerodynamic heating to propel aircraft at hypersonic speeds and the optimization study of low-thrust propulsion systems for space missions. He attended the University of Michigan where he received his B.S. degree in 1944. In 1961, Mr. Moeckel received the Arthur S. Fleming Award, which is given annually to men under 40 who have made outstanding contributions in Federal Government Service. He is an Associate Fellow in the Institute of Aeronautical Sciences, a Senior Member of the American Rocket Society, and a member of the American Physical Society.*

Introduction

By Wolfgang E. Moeckel

The NASA program of research and development in the field of electric propulsion is aimed at eventual use of electric rockets for propulsion of unmanned and manned space vehicles, primarily for interplanetary missions. There are several other promising applications, such as control of the orientation and orbit of Earth satellites and propulsion of lunar supply vehicles, but many mission studies have shown that electric propulsion is most attractive for missions beyond the Moon.

This attractiveness, however, depends on the attainment of certain goals with regard to weight and performance. Perhaps the most critical of these goals is that of developing electric-power generation systems with low enough weight per unit electric power produced and with long enough lifetime. Many of the problems involved in developing such systems have already been discussed, but one of the most difficult, namely, the radiation of waste heat, will be discussed in one of the following papers.

The importance of low specific powerplant weight is illustrated in figures 1 and 2. Figure 1 shows the payload ratio as a function of trip time for scientific probe missions to Mars and to Jupiter. In this payload ratio, the initial weight is that of the electrically propelled vehicle launched into a low Earth orbit; and the payload is the weight, exclusive of electric power, delivered into the Mars or Jupiter orbit. For electrically propelled vehicles, no additional power supply is required for communication and payload operation, so that the actual payload advantage of electrically propelled vehicles is correspondingly enhanced by the amount of weight that would be required to provide adequate nonpropulsive power with other propulsion systems. For many space-probe missions, the availability of large amounts of electric power should prove to be invaluable in increasing the amount and the variety of data that can be accumulated and transmitted in a single mission. Among the possibilities, for example, is continuous television transmission from a mobile vehicle on the surface of Mars or Venus.

Figure 1 shows that reasonable payloads can be carried to Mars and Jupiter with specific weights as high as 50 pounds per kilowatt. These specific weights are in terms of pounds of powerplant weight per kilowatt of power in the

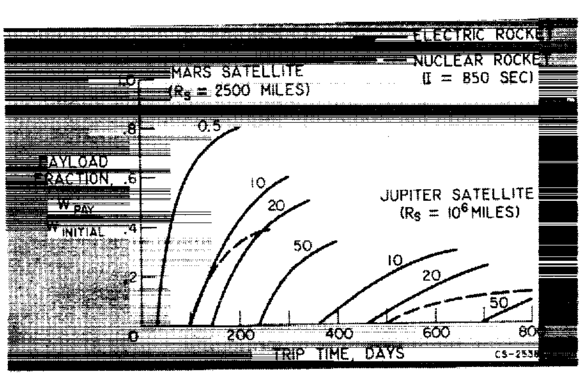

FIGURE 1.—Interplanetary probe missions. (Numbers beside curves are specific weight, lb/kw.)

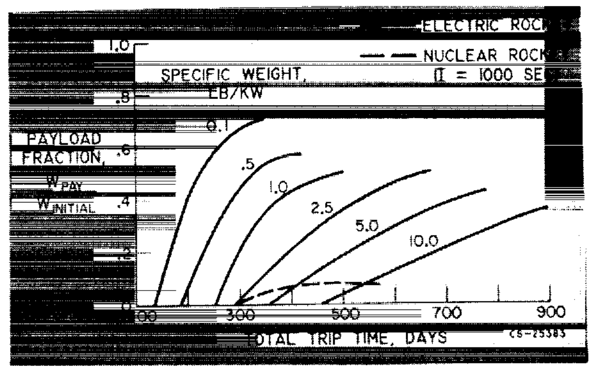

FIGURE 2.—Payload ratio for Mars round trip. Time at Mars, 50 days.

propulsive jet. They therefore include the inefficiencies of the electric rocket (also called thrustor) in converting electric power into jet power. The specific weights of the power generation system (in lb/electric kw) are the values shown in this figure divided by the thrustor power efficiency. The figure shows that large reductions in trip time are possible as specific weight is reduced.

Shown for comparison are values of the payload ratio obtainable with a nuclear rocket with a specific impulse of 850 seconds and a powerplant weight about 10 percent of the initial vehicle weight. The nuclear rocket is competitive with electric propulsion systems having specific weights greater than 10 to 20 pounds per kilowatt for the Mars trip and somewhat higher for the Jupiter mission if the required electric powerplant for the payload and communication is neglected. A similar comparison for the more difficult Mars round-trip mission is shown in figure 2. This mission corresponds approximately to a manned exploration expedition except that the payload in the case shown has been assumed to be fixed, whereas for an actual manned mission, supply consumption en route and the abandonment of the Mars landing vehicle should be considered. These more detailed calculations, however, yield the same relative weights as those given in figure 2, where results are given in terms of ratios that are independent of abolute size. The conclusion to be reached from figure 2 is that manned missions to Mars can be accomplished with greater payload ratios (less initial weight) with electric rockets than with nuclear rockets if powerplant specific weights of the order of 10 pounds per kilowatt can be attained. Nuclear rockets, however, can achieve shorter total mission time unless the specific weight of electric rockets is reduced to 5 pounds per kilowatt or less.

Figures 1 and 2, then, illustrate the goals of electric propulsion systems with regard to specific weight, i.e., less than 50 pounds per kilowatt for interplanetary probes, and less than 10 pounds per kilowatt for the higher-power manned interplanetary missions of the more distant future. The figures also show the very large savings in trip time that are possible if specific weight can be greatly improved. The very low values of specific weight shown in figures 1 and 2 may be considered academic with regard to power generators now under development. Mr. Edmund Callaghan, however, discusses two propulsion systems for which values of specific weight in the range of 1 pound per kilowatt or less appear theoretically possible.

The specific weights discussed are, as mentioned before, the weights per kilowatt of jet power. If the efficiency of the thrustor in converting electric power into jet power is very high, these values are close to those for the weight per kilowatt of electric power generated. An inefficiency in the thrustor effectively degrades the specific weight of the entire propulsion system. Another performance goal in electric propulsion research is therefore high efficiency in conversion of electric power into thrust.

As will be pointed out in subsequent papers, the efficiency of electric thrustors is a strong function of the specific impulse, or jet velocity, produced. Consequently, the range of specific impulse required for contemplated space missions must be known. As shown in table I, this range can be determined roughly from the total impulse required for the mission and the allowable ratio of propellant weight to initial weight. For satellite orientation and orbit control, the total impulse, even for several years of operation, can be quite low, but the propellant ratio must also be low. For the remaining missions,

TABLE I.—*Specific Impulse Range for Electric Propulsion Missions*

Mission (constant-thrust)	Typical total impulse, $a\mathcal{t}$, sec	Typical specific impulse, [a]I, sec
Satellite orbit control	30–100	300–5000
Mars and Venus orbiters	1000–3000	2000–6000
24-Hr satellite orbit	500	1000
Mars and Venus round trips	2500–6000	5000–12,000
Jupiter orbiter (10^6 mile orbit)	3000	6000

[a] $I=\frac{F}{\dot{W}_p}=\frac{Ft/W_o}{W_p/W_o}=\frac{a\mathcal{t}}{W_p/W_o}=\frac{\text{total impulse}}{\text{propellant ratio}}$.

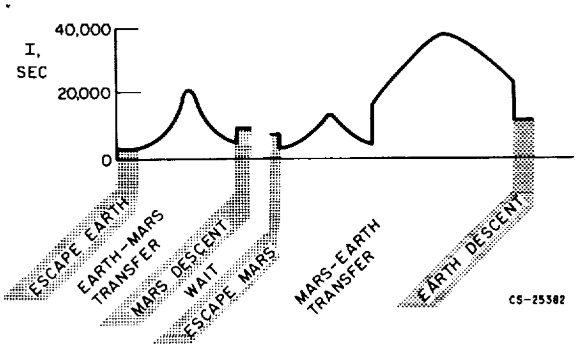

FIGURE 3.—Specific impulse program for constant-power Mars round trip. Total trip time, 500 days; specific powerplant weight, 10 pounds per kilowatt.

where electric rockets are the primary propulsion system, propellant weight ratios in the neighborhood of 0.5 are allowable. Table I shows that, for constant-thrust missions, a large range of specific impulses from below 100 to over 10,000 seconds is needed.

Shown in figure 3 are the specific impulses required during a round-trip Mars mission. These values are for a particular family of trajectories for which the total power is constant, but specific impulse and thrust are allowed to vary. Trajectories of this type are somewhat more efficient with regard to propellant consumption for a given trip time than constant-thrust trajectories. Figure 3 shows that, to follow such trajectories, thrustors that can operate efficiently over a very wide range of specific impulse from 2000 through 40,000 seconds are needed.

Table I and figure 3 show that the further goal of electric propulsion research should be to develop thrustors that can operate with high efficiency over a range of specific impulses from below 1000 to above 40,000 seconds. It is doubtful whether such a range can be covered efficiently with a single type of thrustor. Consequently, research is proceeding on several varieties of electric thrustor, which will be described subsequently.

Another major goal of electric propulsion research is a very long, reliable operating lifetime for all components of the system. The difficulty of achieving this goal for lightweight power-generation equipment was discussed in the Space Power Session. These problems are reviewed and elaborated upon by Mr. Lieblein, particularly with regard to the radiator problem for systems that use a closed thermodynamic cycle. The need for long lifetimes with electric propulsion systems stems mainly from the fact that they are basically low-thrust systems. Since the power delivered by the jet is the product of thrust and jet velocity, it is evident that, for any given power level, thrust decreases as jet velocity or specific impulse increases. Furthermore, for a closed thermodynamic cycle, the weight required to generate a given power level is much greater than for open cycles such as in chemical or nuclear rockets. Consequently, thrust is reduced and weight is increased to achieve the high specific impulses possible with electric propulsion systems. Thus to achieve a given total impulse (thrust multiplied by propulsion time) needed for a mission requires greater propulsion time as thrust is reduced and weight is increased. Typically, for specific weights of the order of 10 pounds per kilowatt at 10,000 seconds specific impulse, the ratio of thrust developed to weight of the powerplant is of the order of 5×10^{-4}. These low thrust-weight ratios require that thrust be applied for months, or almost continuously, throughout an interplanetary mission. Lifetimes of the order of 1 year are therefore a goal in electric propulsion research.

To summarize these goals, extensive use of electric propulsion for space missions requires development of power-generation equipment with specific weights less than 50 pounds per kilowatt for unmanned probe missions and less than 10 pounds per kilowatt for manned interplanetary missions.

Also required are thrustors capable of operating at high efficiency (greater than 70 percent, e.g.) over a range of specific impulses from 1000 to over 10,000 seconds. Also, all components of the electric propulsion system should operate reliably for periods of time of the order of 1 year or longer.

SYMBOLS

a_0	F/W_0
F	thrust, lb
I	specific impulse, sec
R_s	radius of satellite orbit
t	time, sec
W_0	initial vehicle weight, lb
W_p	propellant weight, lb
α	specific powerplant weight, lb/kw

47. Special Requirements on Power Generation Systems for Electric Propulsion

By Seymour Lieblein

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INTRODUCTION

In the preceding paper, Mr. Moeckel describes briefly the long-range electric propulsion missions contemplated by NASA. In the Space Power Session, power generation systems and some of the problems involved in their development for space application are discussed in general. A further discussion of the powerplants required specifically for long-range interplanetary missions in which electric rockets are used is presented herein.

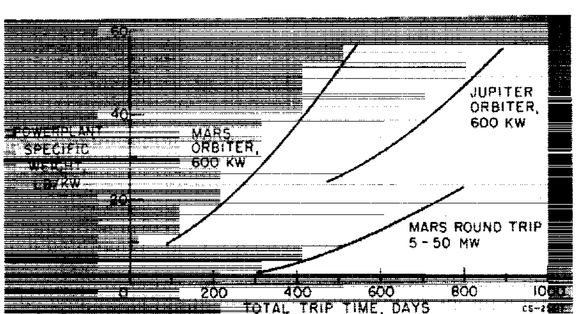


FIGURE 47-1.—Electric powerplant specific weight for equal payload weight for nuclear rocket and electric rocket.

The requirements placed on powerplants for advanced electric propulsion missions can best be summarized as shown in figure 47-1 for the sake of reiteration. Powerplant specific weight in pounds per kilowatt of jet power is plotted against total trip time for equal payloads for electric rockets and nuclear rockets for several missions. Values below each curve indicate the electric rocket to be superior. The lower the power-plant weight, the shorter is the total trip time required for each mission. In particular, extremely low powerplant weights by present standards are required for manned round-trip interplanetary missions with electric rockets.

In summary, power levels from around 600 kilowatts up to 50 megawatts are required; specific weights of the order of 10 pounds per kilowatt or less up to 50 pounds per kilowatt must be achieved; and operating lifetimes from 100 to 800 days have to be considered. In addition, extreme emphasis must be placed upon powerplant reliability to achieve continuous unattended operation for the duration of the mission. In this respect, reduction of powerplant weight is doubly important in that reduced weight leads to shorter trip times, which, in turn, leads to a better chance of achieving the desired reliability.

Electric powerplants suitable for interplanetary missions are discussed, and a brief review of the various problem areas involved in their development is given herein. In particular, a detailed discussion of the problems involved in the dissipation of waste heat, namely the radiator problem, is given with particular reference to the question of protection against impact from meteoroids. It is believed that the radiator currently constitutes the largest obstacle in the achievement of low weight and long-time operation for these powerplants.

POWER GENERATION SYSTEM

General Characteristics

(1) For the long mission times and low specific weights required for interplanetary missions, it is clear that a nuclear reactor power source must be used. Therefore, the powerplant must be started in orbit because of safety considerations.

(2) The systems must operate on closed thermodynamic cycles; cycle efficiencies are therefore governed by Carnot considerations. Furthermore, considerable amounts of waste heat must be continuously rejected, and since thermal radiation is the only mechanism for the rejection of waste heat in space, large radiating surface areas will be required.

(3) In view of the need for high cycle efficiency and low radiator area, high operating temperatures must be utilized. The use of these temperatures results in problems in the selection of materials and working fluids.

To date, the type of powerplant system that appears most attractive for long-range missions is the nuclear turbogenerator power conversion system. As is discussed previously, the nuclear thermionic and magnetohydrodynamic converter systems have also been proposed to achieve the low weights required for advanced missions. These systems contain no moving parts; however, they do need higher operating temperatures in order to produce the same cycle efficiencies as the turbogenerator systems. In addition, since they are closed cycles, they will also require waste-heat removal, and the basic radiator problem remains. In general, the thermionic and magnetohydrodynamic systems have not reached the state of development that the turbogenerator has. Accordingly, attention is centered on the turbogenerator system.

Rankine Cycle Turbogenerator

The basic elements of the turbogenerator system based on a Rankine vapor cycle are illustrated in figure 47–2. Heat from the reactor is supplied to the boiler where working fluid is vaporized. The vapor then drives the turbine, which, in turn, powers the generator producing the desired electric-power output. The vapor exhausting from the turbine is then condensed in a condenser-radiator where its heat of condensation is given up to space by thermal radiation. The condensate is then returned to the circuit through a pump. Inasmuch as condensation of vapor is involved, heat rejection occurs at a constant temperature. For this type of cycle, an alkali metal working fluid, such as potassium, cesium, rubidium, or sodium, is required to keep the fluid pressures at reasonable values for the temperatures involved.

As is indicated previously, the Brayton cycle using an inert gas such as helium or argon has also been proposed in an effort to circumvent the alkali metal corrosion and two-phase flow problems of the Rankine cycle. Since single-phase flow is involved in the gas cycle, however, radiation occurs at a decreasing temperature from radiator inlet to outlet. Radiators for gas cycles, therefore, have a greater surface area

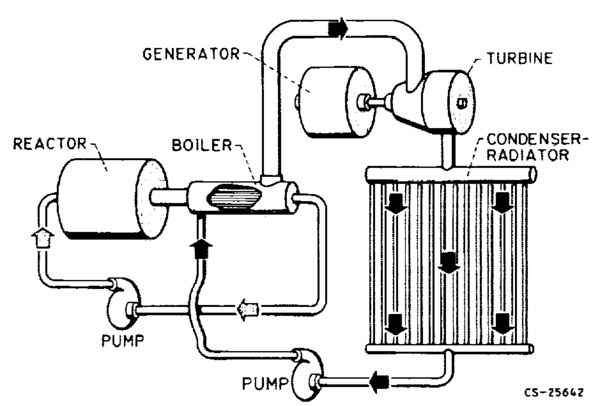
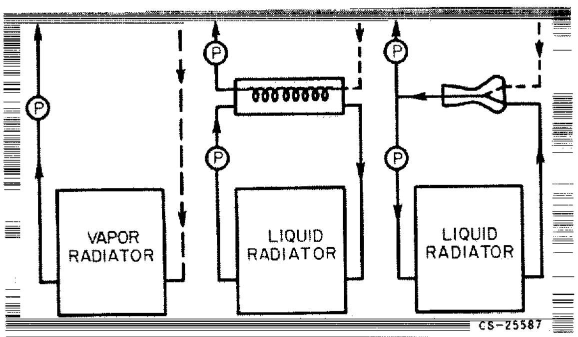


FIGURE 47–2.—Rankine cycle space power system.

requirement than those for the corresponding Rankine cycle. For this and other reasons, the gas cycle will not be considered in this discussion.

For the Rankine vapor cycle, there is a further degree of freedom in how the vapor from the turbine is condensed, as shown in figure 47-3. In figure 47-3(a) is shown the direct condenser in which vapor from the turbine passes directly into the radiator and is condensed in the radiator tubes.

In the second method (fig. 47-3(b)), called the heat-exchanger condenser, a form of heat exchanger similar to the shell and tube type is used to condense the vapor. Subcooled liquid



(a) Direct condenser. (b) Heat-exchanger condenser. (c) Jet condenser.

FIGURE 47-3.—Condensing methods for Rankine cycle.

is provided by a circuit that passes through an all-liquid radiator.

In the third method (fig. 47-3(c)), vapor from the turbine is physically mixed with subcooled liquid, which is obtained from an all-liquid radiator circuit. Part of the mixed condensate is returned to the cycle and the remainder to the liquid radiator loop. There are various advantages and disadvantages associated with these three condensing systems, and they are currently under intensive study.

Radiator Characteristics

As far as radiator characteristics are concerned, typical surface area requirements will be considered first. Figure 47-4 shows specific radiator area in terms of electrical output power against radiator temperature for fixed values of peak cycle temperature. The

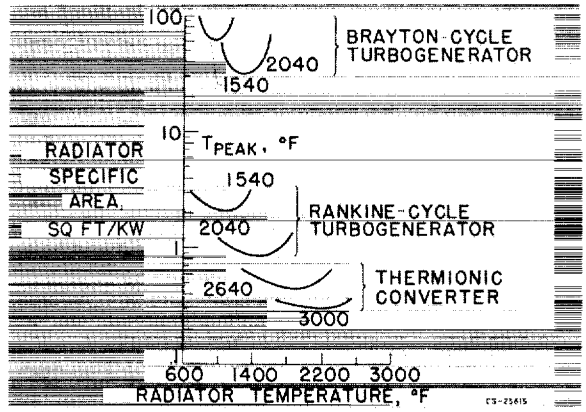


FIGURE 47-4.—Radiator-specific-area requirements for several power-conversion systems.

upper curves are for the Brayton cycle, the center curves are for the Rankine cycle, and the lower curves represent the thermionic converter. Obviously rather substantial radiator areas are required for these systems. For example, for a Rankine cycle turbogenerator, between 1000 and 2000 square feet of isothermal surface area might be required for every megawatt of desired power output. Such large surface areas can result in complex problems for the powerplant system such as structural complications; difficulties in system startup due to large heat capacity; and, finally and most important, high vulnerability to damage from impact with meteoroids.

An example of a radiator configuration currently being considered for electric rocket systems is given in figure 47-5. This type is referred to as the fin-and-tube radiator and is the

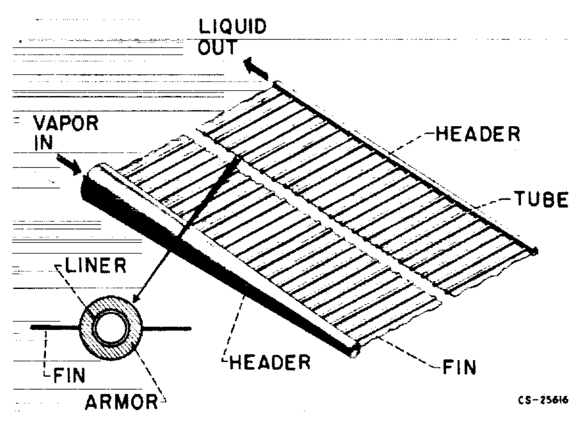


FIGURE 47-5.—Fin-and-tube radiator.

general type being used in low-power-level designs such as Snap 2 and 8. In this configuration for a direct radiator, vapor from the turbine entering the header is distributed to a large number of parallel finned tubes. The vapor condenses within these tubes, and the heat of condensation is radiated to space from the outer surfaces of the tubes and the fins between them. The condensate is then collected in a header and returned to the cycle.

A typical fin-and-tube construction is indicated in the lower part of the figure. The tube is shown to be composed of an inner liner of refractory material to contain the corrosive liquid-metal working fluid. Around the liner is a sleeve of armor sufficiently thick to stop an impacting meteoroid particle. Conducting fins that act as extended heat-transfer surfaces are spaced between the tubes. Radiators for all-liquid circuits will be similar in configuration except that the inlet header and the tube diameters will be smaller.

Fin-and-tube radiators can be used in a large number of configurations for integration with the vehicle. Some are shown in figure 47–6. For the flat-plate type of radiator, means must be provided for packaging the radiator during launch and unfolding before startup in orbit. The cylindrical type can be used if the radiator forms part of the vehicle outer skin; conical shapes can also be used in this respect. Finally, a modification of the flat-plate radiator that can be utilized is the triform configuration. This configuration can also fit within the confines of the launch vehicle skin.

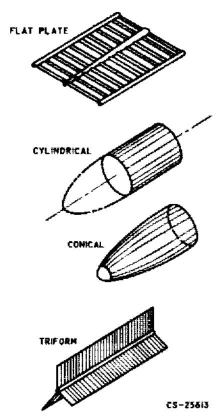


FIGURE 47–6.—Fluid radiator configurations.

Weight Breakdown

Inasmuch as specific weight is the crucial item for interplanetary applications, the weight breakdown for the components of the powerplant system is of great interest. Representative values for the weight breakdown for a 1-megawatt-level Rankine cycle using the tubular radiator configuration illustrated herein are shown in the following table:

Component	Percent of total weight
Reactor and shielding (unmanned)	15–20
Primary loop (pump, boiler, piping)	5–10
Turboalternator and power conditioning	20–25
Heat-rejection system (condenser, unsegmented radiators)	40–50
Other	10–15

It should be noted that the large weight percentage associated with the radiator may also be typical of other closed-cycle power generating systems. Unfortunately, as will be shown later, this largest weight contribution also constitutes the greatest area of uncertainty in current design techniques.

Analysis of tubular radiators indicates that the specific weight involved in the protective armor tends to increase as power level and surface area are increased. Furthermore, most of the weight of the radiator is contained in the protective armor for high-power-level applications (ref. 1). Figure 47–7 shows the variation of tubular radiator weight in pounds per square foot of surface area as a function of sur-

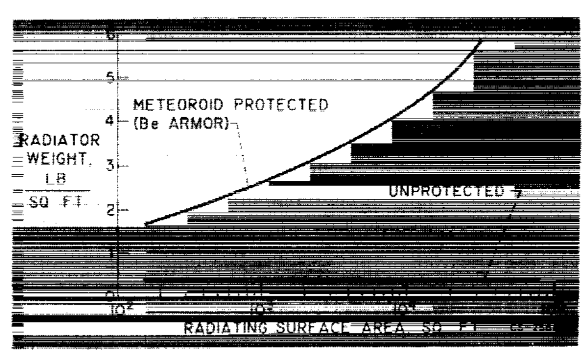


FIGURE 47–7.—Effect of meteoroid protection on radiator weight.

face area for a meteoroid-protected and an unprotected radiator for a typical Rankine cycle. It is seen that, in the megawatt range where thousands of square feet of surface are required, the armor protection can well constitute over three-fourths of the weight of the radiator. It is important, therefore, that the question of the meteoroid hazard and the protection required to defeat it be investigated further.

METEOROID PROTECTION

Present knowledge of meteoroid characteristics (ref. 2) comes primarily from photographic and radar observations of meteors striking the Earth's atmosphere and also from limited data taken from satellites and ballistic-rocket experiments. It is believed that about 80 percent of the meteoroids are sporadic in nature and that only 20 percent appear as showers whose location and time of occurrence are reasonably predictable. More than 90 percent of meteoroids are believed to be of cometary origin with densities ranging from 0.05 to 3.5 grams per cubic centimeter (av., 0.6g/cc). The remaining few percent are asteroidal in origin with densities ranging from 3.5 to 8.0 grams per cubic centimeter. For each category, the density tends to vary with the mass of the meteoroid particle.

Most of the meteoroids lie near the plane of the ecliptic and are in direct orbits similar to that of the Earth. Velocities with respect to the Earth range from 11 to 72 kilometers per second and with respect to a vehicle radiator,

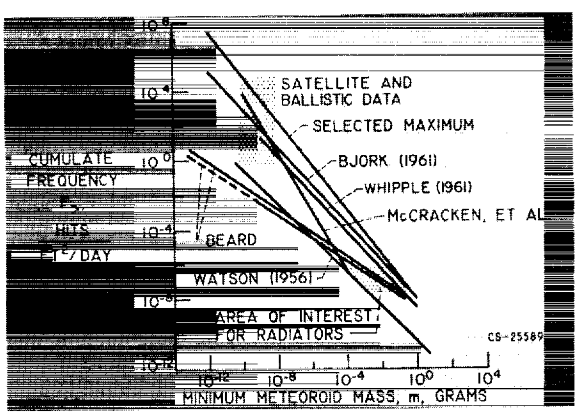


FIGURE 47-8.—Meteoroid mass-frequency distribution.

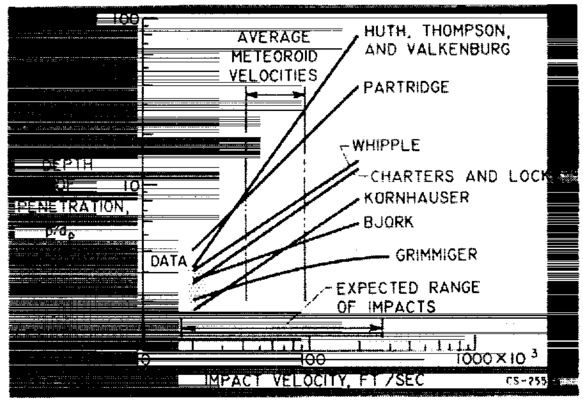


FIGURE 47-9.—Comparison of penetration theories.

from 0 to 84 kilometers per second. Maximum velocities can therefore occur up to around 250,000 feet per second (av., 90,000 ft/sec). The meteoroids vary in structure from a solid to a matrix form and in size from a micron up to several miles in diameter. Fortunately, however, the very large sizes are very infrequent near Earth.

Information on the number of hits likely to be encountered in space is shown in figure 47–8. Here the cumulative frequency in number of hits per square foot of surface per day is plotted as a function of the minimum meteoroid mass in grams. Also shown are various estimates of this mass frequency distribution as obtained by several investigators. The region of data in the upper left was obtained from satellite and ballistic experiments. The dotted area on the right shows the area of interest for large space radiators. It is important to note here that a wide range of frequency distributions is obtained such that there is an uncertainty of several orders of magnitude for the flux distribution in the range of interest.

There is a corresponding uncertainty in the depth of penetration that results from impact with meteoroids as indicated in figure 47–9 as an example. The depth of penetration expressed as a ratio of the penetration depth p to the diameter of the impacting particle d_p is plotted as a function of impact velocity V for several penetration theories as obtained by various investigators (for same target and pellet). Hypervelocity impact data have been obtained in the laboratory, but only at relatively low

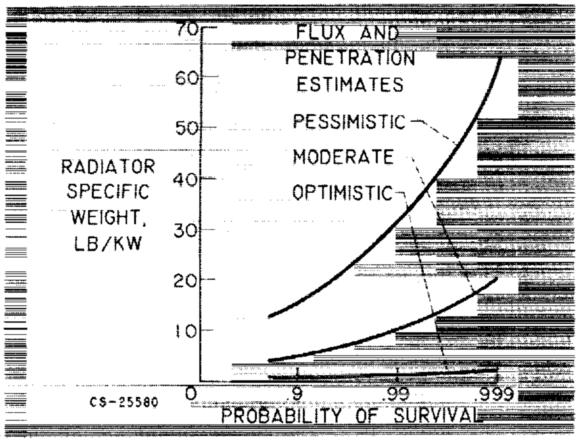


FIGURE 47-10.—Radiator weight estimates for 1-megawatt Rankine cycle.

velocities as shown in the figure. It is clear that there is a wide range of variation in depth of penetration at the expected velocities of meteoroids.

The indicated uncertainties in both the flux distribution and the depth of penetration will be reflected in a corresponding uncertainty in the protection required for space radiators as illustrated in figure 47-10. Here radiator specific weight in pounds per kilowatt of electric power output is plotted as a function of probability of survival. Also plotted are three estimates of weight depending upon whether a pessimistic, optimistic, or moderate (as currently used) interpretation of the available flux and penetration data is used. The curves show that a wide range of estimated weights can be obtained for the required protection. A resolution of the uncertainties involved in the meteoroid hazard to space radiators must therefore be obtained.

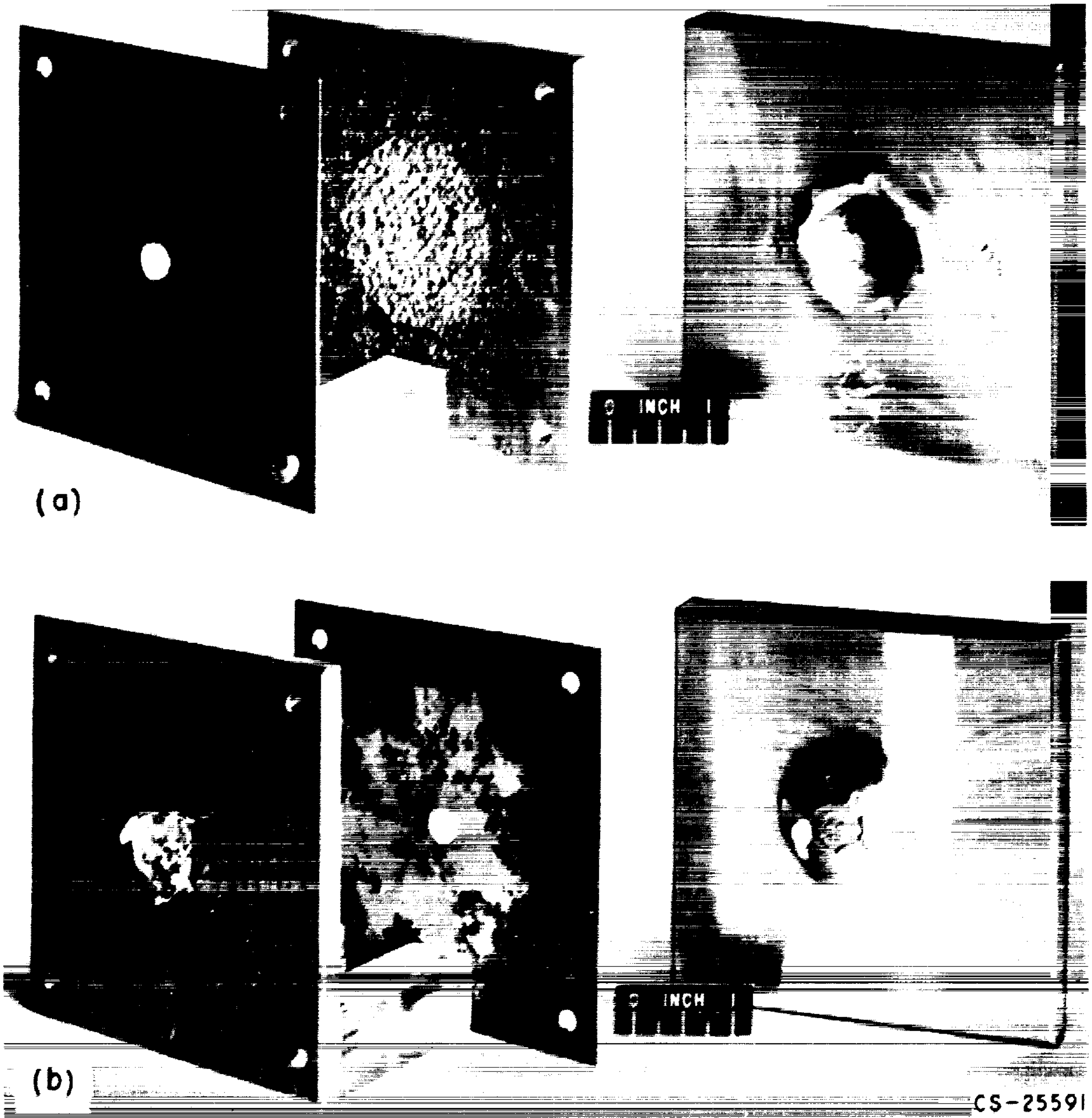
Figure 47-10 represents the degree of uncertainty involved in the required protection against simple puncture from impacting meteoroids. Other damage mechanisms, however, may exist in radiator configurations. For example, spalling can occur on the inside surfaces of the fluid-carrying tubes. The nature of the spalling phenomenon is indicated in figure 47-11. In the upper part of the figure are results of hypervelocity impact into a thick target on the right and into a thin "bumper" plate on the left. On the right is the deep crater resulting from the impact into the thick target; on the left the bumper plate has dispersed the pellet material over a wide area of the back plate. For both plates, however, as shown in figure 47-11(b), even though puncture did not occur, material was spalled or chipped off the back side of the plate. Spalled material in radiator tubes can be just as serious as a puncture. Protection for space radiators may therefore be somewhat more complex than indicated by the simple penetration relations shown earlier.

Since depth of penetration depends on target material, the weight of a fin-tube radiator will depend to a large extent upon the material that is used for the armor protection (ref. 3). This is illustrated in figure 47-12, a plot of radiator specific weight as a function of radiator temperature for a typical Rankine cycle for a 1-megawatt power output at a peak temperature of 2000° F. Meteoroid protection was computed according to a moderate estimate of the meteoroid hazard for a no-puncture probability of 0.9 for 500 days. It can be seen that a wide range of weights can be obtained for different armor materials. Beryllium appears to be the best for producing a minimum weight. Fabrication and bonding problems, however, may exist for this material. It should also be noted that, since beryllium is unusable at temperatures higher than around 1400° F because of strength and sublimation characteristics, the use of higher radiator temperatures will actually involve a large increase in weight if it is necessary to use other armor materials such as stainless steel, columbium, or molybdenum.

APPROACHES TO METEOROID PROBLEM

In view of the critical nature of the meteoroid hazard and the uncertainties involved in designing for it, the designer is faced with adopting various approaches to the problem:

(1) First, of course, he would like a better determination of meteoroid characteristics radiators. For this he would like an intensified determination of meteoroids characteristics from radar and photographic observations from more significant space satellite experiments, and from laboratory hypervelocity firings into actual radiator sections at operating conditions. Efforts along these lines are currently under



(a) Front view. (b) Back view.

FIGURE 47-11.—Impact on soft aluminum targets. Aluminum pellet diameter, 3/16 inch; velocity, 18,500 feet per second.

way, and improved information should be forthcoming.

(2) Radiator area and vulnerability can also be reduced by increasing peak cycle temperatures as shown in figure 47-4. However, if temperatures become very high, it may no longer be possible to use lightweight armor materials such as beryllium, and, as shown previously, no net gain in weight will occur. In addition, the higher temperatures may involve further difficulties with respect to material strength and corrosion.

(3) Various techniques for self-sealing and repairing are possible. In view of the com-

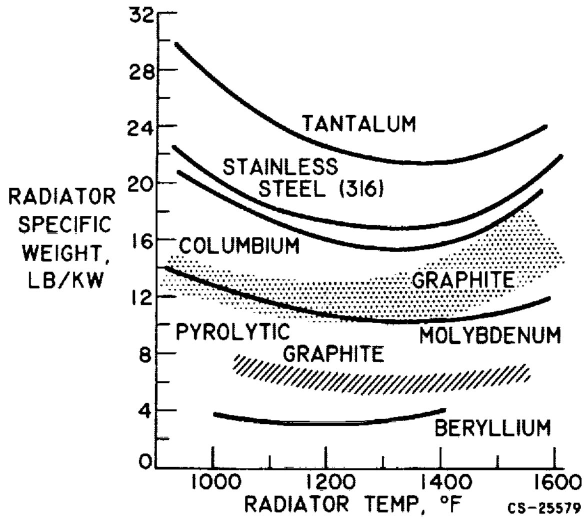


FIGURE 47–12.—Radiator specific weight for various thin and armor materials for a typical 1-megawatt Rankine cycle.

plexities involved in such techniques, however, the prognosis for successful development does not appear optimistic at the moment.

(4) A more promising approach to reducing radiator vulnerability, especially for manned missions where high survival probabilities are required, is in the concept of segmented or redundant radiators. In this technique, the radiator is divided into a large number of independent segments that can be isolated from the rest of the system in the event of a puncture. The theoretical potential for weight saving resulting from the use of this concept (ref. 4) is shown in figure 47–13. Plotted is a weight function against number of isolatable segments for two survival probabilities of 0.9 and 0.999. The calculation assumes that three-fourths of the segments will survive at the end of the mission. As shown by this figure, segmenting can either provide a reduced weight for a given probabiilty or, and this is perhaps more important for a manned mission, it can provide for a large increase in survival probability without an excessive increase in weight, as would be the case for a single-segment radiator. The curves shown in the figure, however, are highly optimistic in that they present only the weights of the radiator tubing. Additional complications and weight penalties will be involved because of the segmenting system, the additional headers, pumps, and other components. Furthermore, the effectiveness of segmenting will also depend on the type of condensing system used, since cutoff valving and leak detection devices may be required.

(5) In another approach the designer can try to determine the best geometric configuration for his radiator to provide the maximum protection for the least weight and complexity. Several fin-tube geometries are illustrated in figure 47–14. On the left are some configurations that embody primarily the armor sleeve approach. The two configurations shown on the right are more typical of the bumper application. In these configurations, the bumper or shield is displaced from the tube surface in order to fragment an impacting particle and spread its energy over a wider area on the tube. A thinner armor sleeve or tube wall can therefore be tolerated. These configurations, however, suffer from the disadvantage that they also present an impedance to the outward flow of heat. Comparative estimates of the relative effectiveness of these geometries are needed.

(6) Another approach to reducing meteroid vulnerability is the use of controlled orientation of the radiator. As was indicated earlier, meteroids are not isotropic in space in that they tend to be concentrated nearer the plane of the ecliptic and in the direction of the Earth's movement around the Sun. It may be possible, therefore, to take advantage of these directional characteristics of meteroids by using a radiator orientation that would result in a lower required armor thickness (ref. 2). The potential gains of controlled orientation are illus-

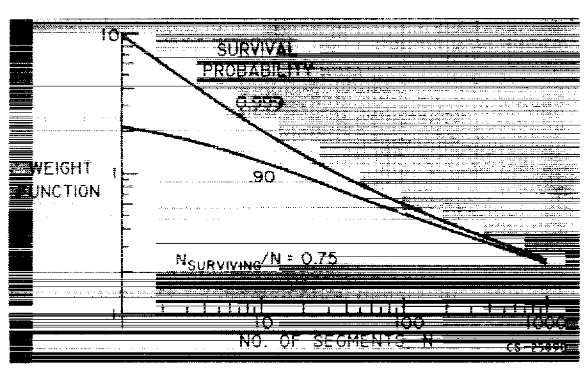


FIGURE 47–13.—Effect of segmenting on radiator panel weight.

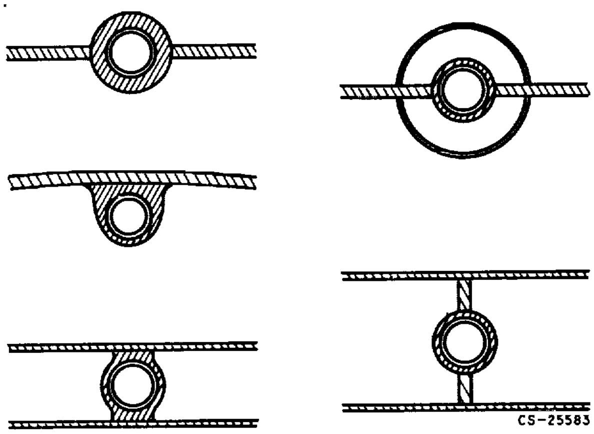


FIGURE 47-14.—Fin-and-tube geometries.

trated in figure 47–15, which is a plot of the ratio of armor thickness required for an oriented radiator t_o to that for an unoriented radiator t_u (isotropic flux) as a function of a design parameter J. The design parameter J involves the meteoroid flux distribution, the variation of depth of penetration with impact velocity, and the angle of impact. Curves are plotted for several radiator orientations sketched in the upper part of the figure. For a plate radiator oriented parallel to the plane of the ecliptic, shown as Case I, a 45-percent reduction in required armor thickness can result in the region of interest.

(7) Finally, there is the potential of producing a low-weight radiator system through the use of a nonfluid type of radiator, which utilizes a moving-belt or rotating-disk concept. In principle, these radiator systems show promise

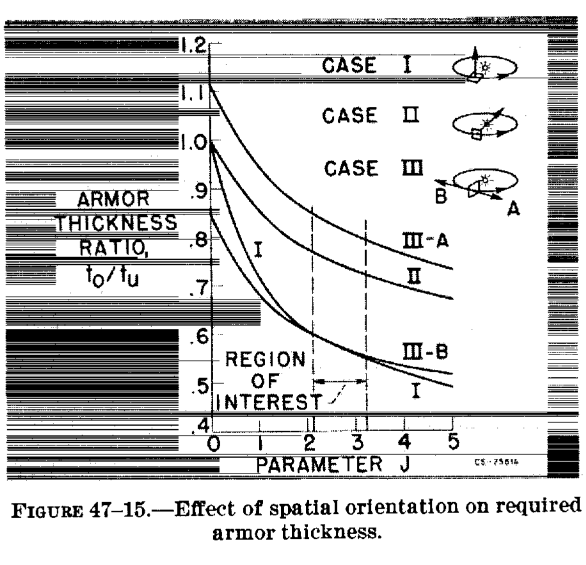
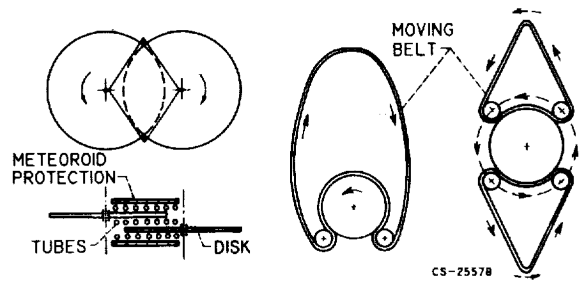


FIGURE 47-15.—Effect of spatial orientation on required armor thickness.

of reduced weight because they present much smaller areas vulnerable to meteroid impact than tubular radiators. Some of the concepts considered in this respect are illustrated in figure 47–16.

Figure 47–16(a) shows the rotating-disk concept (ref. 5) in which rotating disks are placed between banks of tubes and receive their heat by radiation from these tubes. Then, as the disks rotate, the waste heat is radiated to space. Since a large number of rows can be stacked vertically as shown in the figure, meteroid pro-



(a) Rotating disk. (b) Rotating drum. (c) Stationary drum.

FIGURE 47-16.—Nonfluid radiator configurations.

tection will be required only at the ends of the stack, and, therefore, the protection weight can be considerably reduced.

Moving-belt configurations (refs. 6 and 7) are shown in figures 47–16 (b) and (c). In 47–16(b) are a thin moving belt and a rotating drum, which receives the cycle working fluid. Heat is picked up from the drum by means of conduction between the belt and the drum. The heat is then radiated to space as the belt passes through the loop and returns to the drum at a lower temperature. In a second concept (fig. 47–16(c)) the drum can be held stationary, and the belt can be made to rotate around the drum by means of powered rollers. In this version, the principle of heat transmission from the fluid to the drum to the moving belt is the same. The exposed surface area of the drums, which is the vulnerable area in these configurations, will be less than the exposed surface area of a corresponding tubular radiator.

Complex mechanical, structural, and heat-transfer problems are recognized to exist for these nonfluid radiator systems. If the problems inherent in these configurations can be overcome, however, this type of radiator system could be useful.

CONCLUDING REMARKS

This paper has attempted to present some of the special requirements placed upon electric powerplants by the various advanced missions for interplanetary travel with electric rockets. Stringent requirements with respect to powerplant specific weight and long-time reliability were indicated. For closed-cycle systems such as the Rankine turbogenerator, the heaviest system component was estimated to be the waste-heat rejection system. It was also seen that accurate estimates of the weights of suitable electric powerplants are difficult to make because of the large uncertainties involved in the protection required against meteoroid impact for the waste heat radiators. Considerable effort will therefore be required to resolve the uncertainties involved and permit the development of lightweight reliable radiator systems.

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48(a). Generation of Thrust—Electrothermal Thrustors

By John R. Jack

JOHN R. JACK, *Head of the Electrothermal Section of the NASA Lewis Research Center, has conducted research on aerodynamic loads, boundary-layer heat transfer and transition, and electrothermal rockets. Mr. Jack received his B.S. degree in 1946 from Kent State University, and his M.S. degree in 1948 from Carnegie Institute of Technology.*

INTRODUCTION

As Mr. Moeckel noted in his introduction, a specific impulse of the order of 1000 seconds is adequate for several space missions within the gravitational field of the Earth. This specific impulse can be achieved readily with fairly good efficiency by an electrothermal thrust generator. In such a device the propellant is heated electrically before being expanded in a conventional convergent-divergent nozzle.

The effort at the Lewis Research Center in the electrothermal propulsion field is in three areas: (1) the investigation of complete engine configurations, (2) propellant heat-transfer effects, and (3) the investigation of suitable propellants.

ENGINE CONFIGURATIONS

There are two electrothermal propulsion schemes under investigation. The first of these approaches is shown schematically in figure 48(a)-1. This experimental arc jet consists of a cathode and an anode, which also serves as a convergent-divergent nozzle. In operation an arc is struck between the two electrodes. The propellant flows through and around the arc, where it is heated to very high temperatures, and is then expanded through the nozzle to produce thrust.

Arc jet research is being conducted at various research establishments on both alternating- and direct-current units. Currently a 30-kilo-

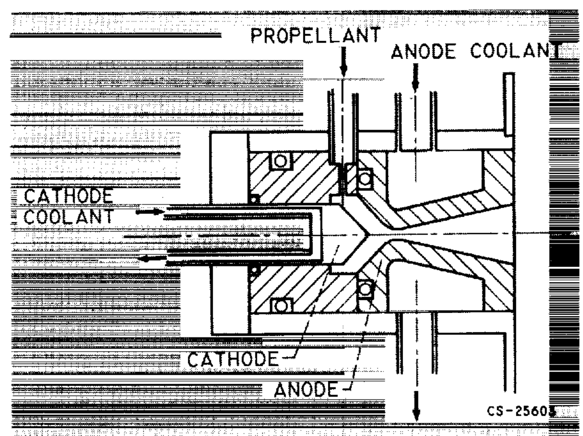

FIGURE 48(a)-1.—Arc-heated thrust device.

watt unit using hydrogen yields a specific impulse of about 1000 seconds, a thrust of ½ pound, and efficiencies of approximately 40 percent. The main problem areas associated with this type of thrustor are: (1) the choice of a suitable propellant to increase efficiency and to meet space storage requirements, (2) the demonstration of electrode life to meet lifetime requirements, and (3) the development of better regenerative and radiation cooling techniques.

The second electrothermal device of interest is shown in figure 48(a)-2. It is an experimental resistance-heated hydrogen rocket designed for a thrust of 1 pound and a specific impulse of 1000 seconds. Its principle of operation is based upon using an electrically powered, resistance-heated heat exchanger to bring

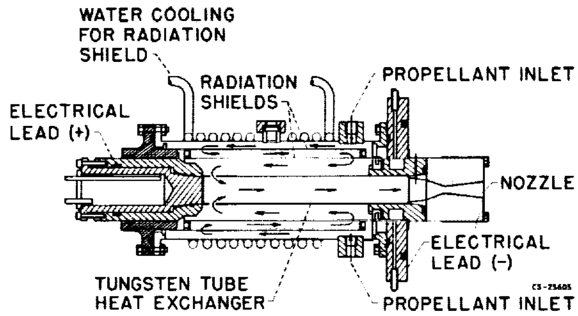

FIGURE 48(a)-2.—30-Kilowatt tungsten tube electrothermal propulsion engine.

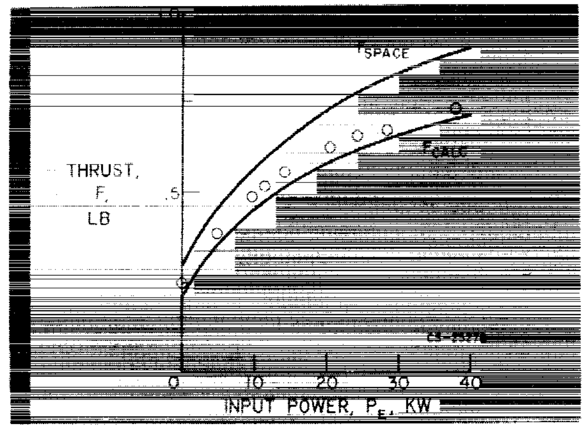

FIGURE 48(a)-3.—Variation of thrust with input power. Propellant weight flow, 10^{-3} pound per second.

the propellant up to the temperature required to produce the desired engine performance. A detailed discussion of this engine may be found in references 1 and 2.

This propulsion approach offers several potentially attractive features. Among them are:

(1) High efficiency

(2) Long life and good reliability

(3) Simple matching to a power supply (this device can operate equally well on either alternating or direct current)

(4) Simple starting technique

(5) Variable thrust

There are two disadvantages associated with this engine. First, since its operation requires a metal heat exchanger, it is temperature limited; consequently, the specific impulse is limited to approximately 1100 seconds. Second, this engine must use hydrogen for a propellant to achieve a specific impulse of 1000 seconds; therefore, the space storage of hydrogen is a problem.

A typical set of experimental thrust data for a resistance-heated hydrogen jet is shown in figure 48(a)-3 for a propellant flow rate of 10^{-3} pound per second. The thrust increases from a cold-flow value of 0.25 pound to a value of 0.73 pound at an input power of 38 kilowatts. Since the propellant flow rate is 10^{-3} pound per second, the specific impulse increases from 250 to 730 seconds. Also presented in figure 51(a)-3 is the calculated thrust based upon one-dimensional gas-flow theory for the gas stagnation temperature at a given power input and nozzle geometry. Experiment and theory are in good agreement. The amount of thrust to be expected in a space environment is also presented. At the highest input power, the thrust to be expected for space operation is 0.9 pound, and the corresponding specific impulse 900 seconds. The vacuum specific impulse of 900 seconds should not be considered an upper limit because this value is based upon a heat-exchanger temperature of 4600° R and a gas stagnation temperature of 4100° R. Both of these temperatures can be increased so that a vacuum specific impulse of 1000 seconds may be obtained.

The flow through the engine is quite uniform, so that studies of a more fundamental nature may be made using the engine as a source of high-temperature hydrogen.

PROPELLANT HEAT-TRANSFER EFFECTS

Two propellant characteristics affect the performance of an electrothermal thrust generator. The first is heat loss to the engine walls by convection. This characteristic is very important because it affects engine performance and ultimately limits thrustor performance. This limitation arises for regenerative cooling because the heat capacity of a propellant at a given engine wall temperature is limited, and, for radiation cooling, because the engine wall material has an operating temperature limit.

A preliminary analysis of this problem has been made for complete regenerative and radiation cooling utilizing hydrogen and helium as propellants (ref. 3). The results for regenerative cooling with hydrogen are shown in figure 48(a)-4. The maximum specific impulse ob-

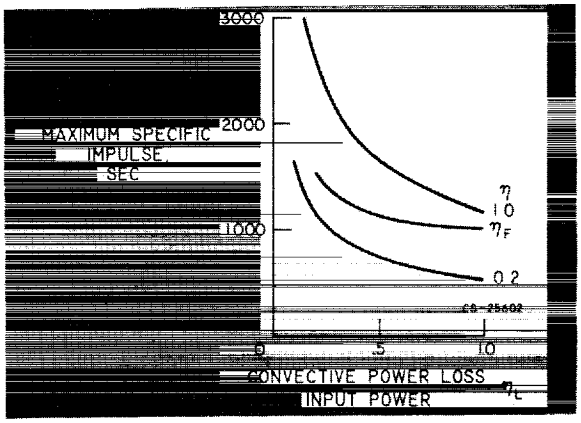

FIGURE 48(a)-4.—Maximum specific impulse with regenerative cooling. Propellant, hydrogen; pressure, 1 atmosphere; wall temperature, 5400° R.

tainable is presented as a function of the amount of heat or power to be recovered regeneratively. Results are presented for three overall engine efficiencies: $\eta=0.2$, a low experimental efficiency; $\eta=1.0$, the maximum engine efficiency; and $\eta=\eta_F$, the maximum efficiency to expect if the flow is frozen (η_F is the frozen flow efficiency and will be discussed in a subsequent section). If the flow is frozen and a typical power loss ratio of $\eta_L=0.2$ is used, it is found that the maximum specific impulse obtainable with regenerative cooling is of the order of 1500 seconds.

Figure 48(a)-5 illustrates what is expected for radiation cooling with hydrogen as a propellant. In this case, the maximum specific impulse is a function of the thrust level, so two typical thrusts have been used, 0.1 and 1.0 pound. The radiating areas used are estimated values appropriate for the thrust levels considered. Assuming again a power loss ratio of $\eta_L=0.2$, the maximum specific impulse attainable at a thrust level of 1.0 pound is approximately 1100 seconds, whereas that found for a thrust level of 0.1 pound is about 1500 seconds. Thus radiation cooling appears more attractive at the lower thrust levels.

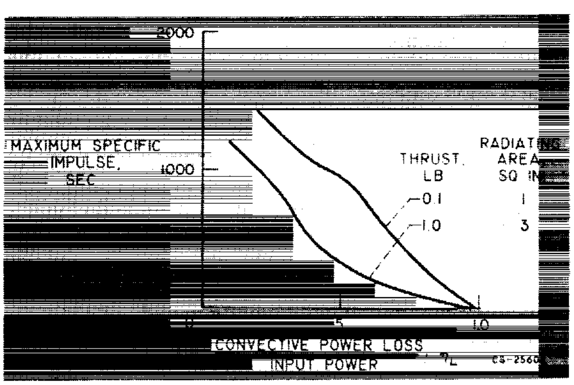

FIGURE 48(a)-5.—Maximum specific impulse with radiation cooling. Propellant, hydrogen; pressure, 1 atmosphere; wall temperature, 5400° R; $\eta=\eta_F(1-\eta_L)$.

SUITABLE PROPELLANTS

As noted previously, hydrogen presents a severe space storage problem, and it is desirable to search for a propellant that yields comparable performance and yet is easily stored. Possible propellants for electrothermal thrustors may be evaluated by considering the second propellant characteristic affecting engine performance. Because of the high temperatures encountered, the propellant dissociates and ionizes, and, if it does not recombine in the nozzle, an energy loss is experienced which decreases the engine performance. Experience to date has indicated that little recombination can be expected, so that the propellant flow can be considered frozen. As a result, the ideal engine efficiency is equal to the frozen-flow efficiency, and propellants may be compared in terms of this parameter. The frozen-flow efficiency is defined as the ratio of the power available for thrust to the total power imparted to the propellant; it is also a measure of the amount of power invested in dissociation and ionization.

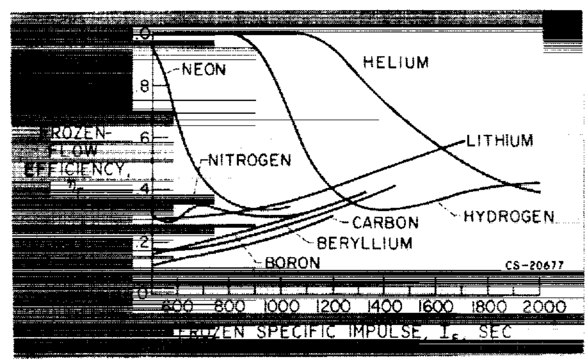

FIGURE 48(a)-6.—Frozen flow efficiencies for various propellants. Pressure, 1 atmosphere.

Figure 48(a)-6 shows a typical comparison obtained from an analytical study (ref. 1). The frozen-flow efficiencies for eight possible propellants are presented as a function of specific impulse for a pressure level of 1 atmosphere. At a specific impulse of 1000 seconds, helium and hydrogen have considerably better efficiencies than any of the other propellants considered. At specific impulses greater than 1600 seconds, lithium is better than either helium or hydrogen.

Although the better propellants discussed here have good efficiency in the specific impulse range of interest, each has associated with it space storage or feed problems; it is thus desirable to look further at other possible propellants. Some possibilities are water, ammonia, ethane, methane, and lithium hydride. These propellants have been studied, and a comparison of frozen-flow efficiencies is shown in figure 48(a)-7.

For specific impulses of about 1000 seconds, hydrogen still yields a better efficiency than any of the noncryogenic propellants considered. It appears, however, that both lithium hydride and water may possibly yield a comparable efficiency and should be given more consideration. For specific impulses of the order of 1200 seconds, both lithium and ammonia are better than hydrogen, and ethane and methane seem promising.

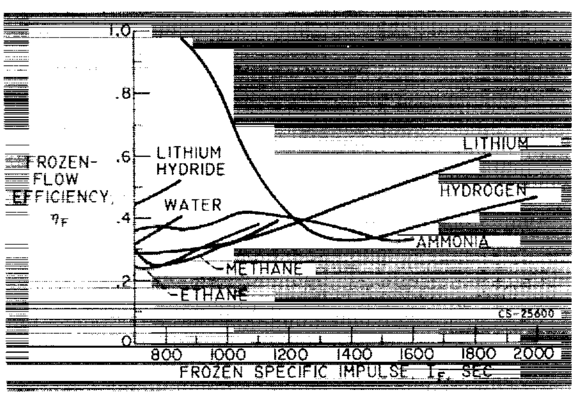

FIGURE 48(a)-7.—Frozen flow efficiencies for several noncryogenic propellants. Pressure, 1 atmosphere.

CONCLUDING REMARKS

During the past few years the research conducted on electrothermal thrustors has narrowed considerably the gap existing between the research model and the flyable prototype. The lifetime of engine components, initially measured in minutes, has now been extended to weeks. Specific impulses, originally measured at a few hundred seconds, have been pushed up into the 1000- to 1500-second range. In fact, it now appears feasible to design a thrustor having the desired performance characteristics and a fairly good conversion efficiency.

Despite the many advances made, however, there still remain several major problem areas that require additional research effort before the overall capability of an electrothermal thrustor can be fully realized. For example, electrodes and heat exchangers that are reliable and can meet mission lifetime requirements must be demonstrated. Noncryogenic propellants that indicate the possibility of yielding good performance and yet could be easily stored in space should be given considerable attention. In conjunction with a propellant investigation, the local convective heating rates should be measured because these will determine the type and characteristics of the cooling system finally employed. Continued research in these areas should ultimately prove very profitable and lead to the development of an efficient and reliable electrothermal thrust unit.

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48(b). Generation of Thrust—Electromagnetic Thrustors

By George R. Seikel

GEORGE R. SEIKEL, *Head of the Plasma Flow Section of the Lewis Research Center, is currently conducting research on the fundamentals of flowing plasma and the possibility of utilizing plasma thrustors for interplanetary electric propulsion systems. He received his B.S. degree in Aeronautical Engineering in 1955 and his M.S. degree in Engineering Mechanics in 1957 from the University of Notre Dame. In September 1961, Mr. Seikel was a delegate to the International Atomic Energy Agency Conference on Plasma Physics and Controlled Nuclear Fusion Research conducted in Salzburg, Austria.*

INTRODUCTION

The general principle of acceleration in all electromagnetic thrustors is essentially the same as that in ordinary electric motors. As illustrated in figure 48(b)-1, if in an electric conductor, be it a wire or a plasma, a current $\vec{j}$ flows in the presence of a magnetic field $\vec{B}$, an electromagnetic body force is produced on the conductor. The body force $\vec{F}$ is equal to the vector or cross product of the current and the magnetic field. The direction of the force is perpendicular to both $\vec{j}$ and $\vec{B}$, as illustrated by the right-hand rule. In electromagnetic thrustors, it is this $\vec{j}\times\vec{B}$ body force that is used to act on the plasma propellant.

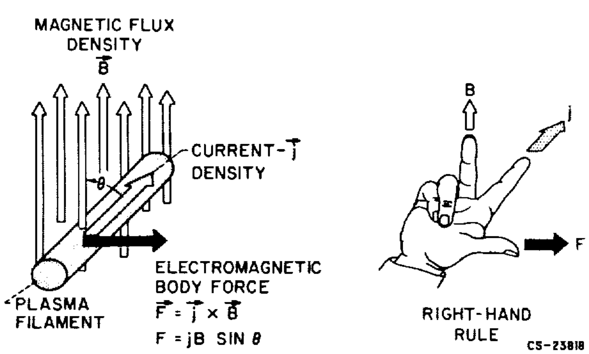

FIGURE 48(b)-1.—Electromagnetic body force on a plasma current filament.

PLASMA PROPELLANT

The general properties of the plasma propellant are as follows:

(1) The plasma is composed of three constituents: electrons, ions, and neutrals. This distinction is exceedingly important since most of the physics is involved with the simultaneous dealing with three different things. In fact, a great deal of confusion can often be eliminated if the species can be individually treated.

(2) The plasma charged particle density is sufficiently high that approximate charge neutrality exists throughout the fluid, except in thin sheaths at all boundaries. Large departures from charge neutrality can exist only over a distance termed the Debye length. The Debye length is, however, always much less than the accelerator characteristic length. The sheaths at the boundaries have thicknesses of the order of a Debye length.

(3) Nonequilibrium effects predominate in the plasma. The mere fact that large currents flow in the plasma guarantees velocity nonequilibrium, for a current requires that electrons are moving relative to ions. Electrons with velocities relative to the ions of one-thousandth of the speed of light would not be unusual. Similarly, ion slip may exist;

that is, an insufficient number of collisions may be present to prevent the ions from slipping through the neutrals. With regard to thermal equilibrium, to expect to be able to do more than assign different temperatures to each of the plasma constituents is usually unreasonable. Even a constituent may not be in self-equilibrium; that is, its particles do not have a Maxwellian particle distribution. Typically, a plasma propellant may have an electron temperature of 100,000° K, while the ions and neutrals are essentially at room temperature.

(4) By metallic standards, the ordinary or scalar conductivity of the plasma propellant is low. Even for a fully ionized plasma, an electron temperature of 5,000,000° K is required to obtain a conductivity equal to that of copper. Typical plasma propellants have electron temperatures of less than a few hundred thousand °K and, thus, have conductivities slightly less than that of carbon, which is used to make resistors. Because of the large energy required to elevate the electron temperature, however, utilizing these low-conductivity plasmas for propulsion applications is desirable.

(5) Hall effects predominate in the accelerators; that is, the currents and the electric fields in the plasma are not parallel. This is a necessary consequence of efficient accelerator operation, since the ratio of the useful work to the Joule heating is either less than, or of the order of, the Hall parameter $\omega_{c-}\tau_{-}$ (where ω_{c-} is the electron cyclotron frequency eB/m_{-}, and τ_{-} is the mean time for an electron to lose its momentum to the ions or neutrals).

PLASMA ACCELERATION

There are theoretically two methods of efficiently accelerating a plasma. One method is to heat the plasma and convert the random energy of the hot propellant into kinetic energy in a nozzle. If a physical nozzle is employed, it is a pure electrothermal device of the general class described by Mr. John R. Jack in the previous paper. If a magnetic nozzle is employed, the accelerator may be classed as an indirect electromagnetic device; that is, no work is done with the electromagnetic body force, but it is utilized to transfer the forces between the plasma and the accelerator structure. The alternative method of accelerating the plasma is to use directly the electromagnetic body force $\vec{j}\times\vec{B}$ to add kinetic energy to the plasma.

Consider the plasma acceleration process from a multifluid point of view; that is, what is happening to the electrons, the ions, and the neutrals? Note that the electrons have more than enough thermal energy to be expelled at any specific impulse of interest to electric propulsion, and the primary method of accelerating neutrals is by collisions with ions. Thus, in any plasma acceleration process the chief concern is the mechanism for accelerating the ion plasma constituent.

TYPES OF ELECTROMAGNETIC ACCELERATORS

A large number of devices use the electromagnetic body force to accelerate a plasma propellant; one convenient method of classification is on the basis of the type of electric power utilized—direct current, alternating current, radio-frequency, or pulsed. Such a division is descriptive and permits estimation of the required power conditioning systems. Power sources presently being developed utilize alternators to generate the electric power. Thus, the minimum power conditioning requirement for d–c accelerators is that the power be rectified; a–c accelerators may be able to operate directly from the alternators. Estimates indicate that up to 100-kilocycle systems appear possible without serious penalties in alternator weight or efficiency. Radio frequency systems would require both rectification and vacuum tubes to condition the power and, thus, have competitive disadvantages in power-conditioning-system efficiency and weight. The pulsed systems require efficient charging circuits and capacitive power storage. Recent advances in capacitor technology indicate that sufficiently light capacitors appear imminent.

A cursory description of some of the plasma accelerator configurations presently being investigated by the Lewis Research Center is given and references to some other devices are cited in the following section. Additional discussion is presented in the paper by Mr. Macon C. Ellis, Jr.

D-C Accelerators

The type of d-c accelerators under investigation range from the direct relatively low-impedance segmented crossed-field accelerators (refs. 1 to 4), to the relatively high-impedance Hall current accelerators (refs. 5 to 10), to the indirect nonequilibrium magnetically contained electrothermal accelerator (refs. 11 and 12).

Hall current ion accelerator.—Figure 48(b)-2 schematically depicts the Hall current ion accelerator (ref. 8), which is annular in geometry. The applied electric field is utilized to accelerate directly the ions produced by the discharges. The applied radial magnetic field is sufficiently weak that it does not affect the ion motion, but is sufficiently strong to be dominant in determining the electron motion. In such a situation the current of the electrons diffusing upstream in the accelerator is substantially less than the azimuthal electron drift or Hall current. Since it is this Hall current that appears to provide the electromagnetic acceleration force, and only the much smaller diffusion electron current and ion current are supplied by the power source, the device has a relatively high impedance. This device is analogous to an ion engine with no space-charge limitations on the ion flux. Since only the ion plasma constituent is accelerated, the accelerator cathode also serves as a neutralizer. Only very preliminary performance data have been obtained for such devices, but the results are promising.

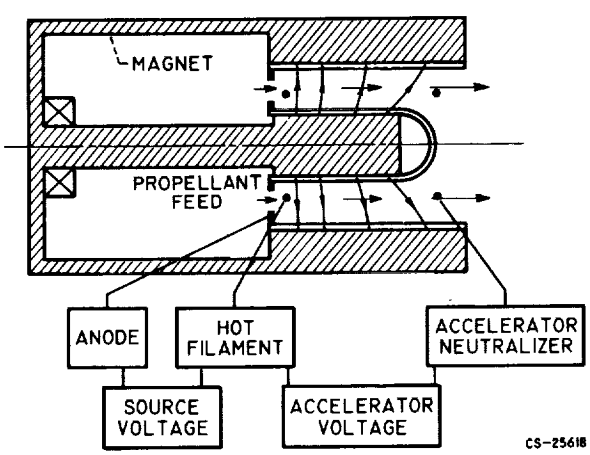

FIGURE 48(b)-2.—Hall current ion accelerator.

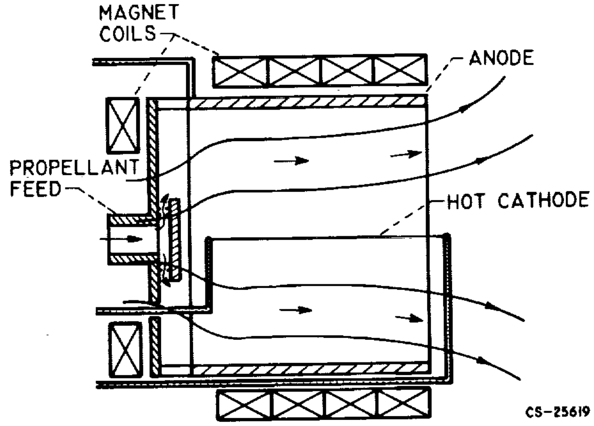

FIGURE 48(b)-3.—Nonequilibrium magnetically contained electrothermal thrustor.

Nonequilibrium electrothermal magnetically contained accelerator.—Figure 48(b)-3 schematically depicts a nonequilibrium electrothermal magnetically contained accelerator (ref. 12). It consists of a heated filament, a cylindrical anode, and a varying axial magnetic field. In low-density discharges of this type, the electric power is directly added to the random energy of the plasma electrons. The high-energy electrons ionize the propellant, and, as the plasma's electron gas expands out of the device, its random electron energy is converted to direct energy. Since there can be no divergence of current, however, the electrons drag the ions along. Physically, this is accomplished by an axial electric field that the plasma itself builds. This field retards the electron's expansion, accelerates the ions, and causes an azimuthal electron Hall current to flow that provides the electromagnetic reaction force on the accelerator. The energy added to the ions is at the expense of the random electron energy. The expansion process is controlled by the magnetic nozzle action of the spatially varying magnetic field. Preliminary performance of a 200-watt thrustor with argon as the propellant has yielded power efficiencies of the order of 20 percent at a specific impulse of 1500 seconds.

A-C Accelerators

All d-c accelerators require that electrodes be in contact with the plasma. This possible limitation of life can be circumvented by a-c electrodeless accelerators that utilize induction

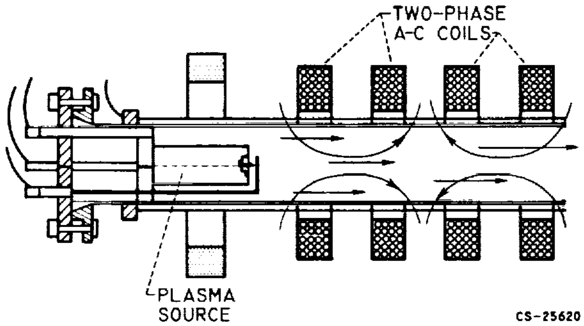

FIGURE 48(b)-4.—Traveling magnetic wave plasma engine.

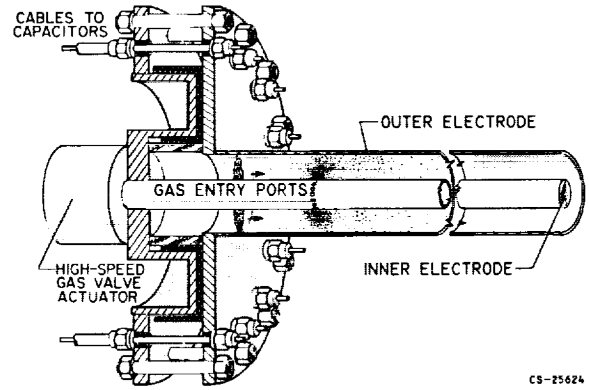

FIGURE 48(b)-5.—Coaxial plasma gun.

coupling; however, the coupling efficiency must be good. A number of electrodeless accelerators are being investigated (refs. 13 to 18). One such device, the traveling magnetic wave accelerator (ref. 16), is schematically depicted in figure 48(b)-4. The plasma produced in the plasma source diffuses into the accelerator and is accelerated by the moving magnetic wave produced by the polyphase coil system. The operation is analogous to a polyphase induction motor. As the magnetic wave moves through the plasma, an azimuthal current is induced. The interaction of this current with the magnetic wave produces an electromagnetic body force tending to drag the plasma with the wave. The body force acts on the plasma's electrons, which transfer the force to the ions through an induced axial polarization or Hall electric field (ref. 10).

Pulsed Accelerators

The efficiency of plasma accelerators tends to increase with power level. The pulsed accelerators, or guns (refs. 19 to 23), attempt to capitalize on this fact by utilizing instantaneous power levels above 100 megawatts. Final versions of such thrustors would have pulsing rates of 100 to 1000 pulses per second and pulse durations of a few microseconds. The chief problem in such systems is in obtaining a sufficiently tight coupling to the gun. Useful power addition to the plasma terminates in less than the first half cycle of the discharge. This places stringent requirements on obtaining low-inductance capacitors and low-inductance circuits at breakdown. An experiment (ref. 23) on one of the most interesting of these accelerators, the coaxial plasma gun, is schematically depicted in figure 48(b)-5. In this gun the low-inductance capacitor bank is discharged before the coaxial pair of electrodes. The radial current sheet interacts with the azimuthal magnetic field created by the current flowing in the electrodes to provide the electromagnetic acceleration force. This force acts on the electrons, which are coupled to the ions by the induced axial-electric field (ref. 24). Typical experiments utilize pulsed propellant injection, but in final systems the pulsing rate may be adequate to permit continuous propellant flow. Experiments may or may not use switching between the capacitors and the gun. Such devices are polarity sensitive, and operation is superior if the center electrode is the initial cathode (ref. 25). Performance to date of such a gun has yielded efficiencies as high as 30 percent at a specific impulse of 5000 seconds.

CONCLUSIONS

In conclusion, although many interesting and attractive schemes are being investigated for electromagnetic propulsion applications, none of these devices has as yet demonstrated the performance required for an actual propulsion mission. On the other hand, no fundamental obstacles have been discovered that would prevent the eventual achievement of this goal. The chief problems in the development of such devices have primarily been the lack of fundamental knowledge of plasma physics and the lack of sufficient plasma diagnostic techniques to evaluate adequately the experiments being performed. Thus, it is actually in these areas that the major effort has been devoted, and in which substantial contributions are being made.

Potentially, electromagnetic accelerators should be able to provide rugged, relatively small thrustors capable of above megawatt power levels. Efficiencies should be competitively attractive, at least in the lower specific impulse range of from 1000 to 5000 seconds. Potentially high efficiency in this range of specific impulses is due to the possibility of efficient utilization of the plasma's ions. Since in such devices the ionization and acceleration processes can be integral, the ions can be accelerated as they are produced.

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48(c). Generation of Thrust—Electrostatic Thrustors

By Warren D. Rayle

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INTRODUCTION

The electrostatic thrustor, as exemplified by the ion rocket, is not a new concept. For example, Oberth (ref. 1) discussed such a system in 1929. A fair amount of theoretical examination had been given to the idea well in advance of the first space flights. Not, however, until shortly before those flights, or the brink of the space age, did the serious development of the associated technology commence. Under the auspices of the Department of Defense as well as NASA, this development has now been carried to the point where prototype devices that might be suitable for some space applications are available.

This discussion has as its purpose (1) to outline the fundamental processes integral to electrostatic propulsion, (2) to describe the main categories of such thrustors and their current status, and (3) to point out a few possible future developments which might change the present emphasis in electric propulsion research. This paper is not a comprehensive survey of the field. The reader seeking more detailed and comprehensive information is directed to the existing literature, especially references 2 to 5.

ELECTROSTATIC PROPULSION FUNDAMENTALS

Figure 48(c)-1 is a schematic diagram of an electrostatic thrustor. The three processes integral to the cycle are (1) the generation of charged particles, (2) their acceleration, and

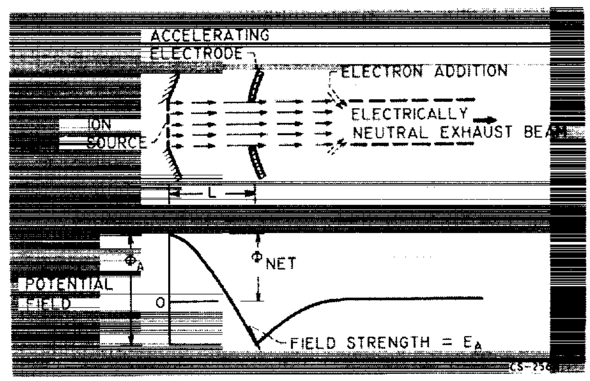

FIGURE 48(c)-1.—Electrostatic thrustor.

(3) the neutralization of the resultant beam. As shown, the ions are accelerated beyond the velocity desired and then decelerated. This process permits greater ion current densities and also prevents electrons from being drawn back into the ion source. In the development of an electrostatic thrustor, the primary goals are high efficiency, thrust-to-weight ratio, and durability. The thrust-to-weight ratio is directly dependent on the attainable thrust density, that is, the thrust per unit area. The maximum current density which an electrostatic accelerator will accept is calculable and is implied in figure 48(c)-1 by the potential gradients being shown as zero at the source. Such limiting currents have been calculated for a wide variety of configurations (ref. 6). The maximum thrust density has also been shown to depend on the electric field at the accelerator

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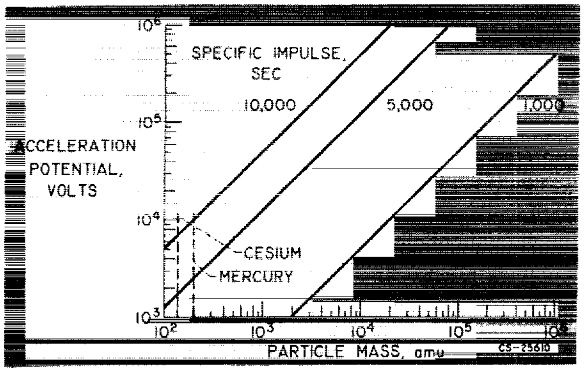

FIGURE 48(c)-2.—Singly charged particle acceleration.

(refs. 2 and 7), which, in turn, is approximated by the ratio of the accelerating potential to the acceleration length.

The net accelerating potential required to obtain a given specific impulse varies, as shown in figure 48(c)-2, with the particle mass. Singly charged particles are assumed. A convenient number to remember is 10 volts per atomic mass unit, which corresponds to a specific impulse of about 4500 seconds. For particles such as cesium or mercury ions, the required accelerating potentials are then on the order of 1000 to 2000 volts. With such a small accelerating potential, acceleration lengths of less than 1 millimeter would be quite feasible if electric breakdown were the only limiting factor. Other difficulties are encountered in attempting to fabricate and operate such a delicate structure. A major consideration is the probable operating life.

An alternative approach to obtaining high thrust densities is through the use of heavier particles. A 100,000 atomic mass unit particle requiring 1 megavolt of acceleration permits accelerating lengths between 0.1 and 1 meter.

The accelerator limits are thus seen to be rather clearly delineated. Ion or charged-particle sources are currently of two principal varieties. One source is known as contact ionization and depends on the fact that some of the alkali metals, notably cesium, have ionization potentials so low that they may lose an electron on contact with a high-work-function surface such as tungsten. Contact ionization would be an extremely efficient method for the production of ions if it were not for the requirement that the tungsten be kept hot in order to prevent the accumulation of a layer of cesium and the consequent reduction of the work function of the surface.

The second system involves the bombardment of the propellant atoms with electrons. This system is not specific to any single propellant, but is in theory able to produce ions of many different materials. The efficiency of this system is a function of many factors but has been shown to be relatively attractive for fairly heavy ions.

The neutralization of the charged-particle beam is a subject to which considerable analytical and experimental effort has been devoted. It is easily demonstrated that neutralization of ion beams can be accomplished inside vacuum tanks. Less easy is the task of proving conclusively that a similar result can be obtained in space where the beam is semi-infinite. Some mechanism must be postulated whereby the mean or drift velocities of the ions and electrons can be synchronized. Analysis is continuing and, in addition, an experiment is being readied whereby actual operation of two types of ion engine will be attempted in space.

PRESENT STATUS OF THREE TYPES OF THRUSTOR

Ion engines in the operational prototype stage include the contact engine in which cesium is used as the propellant and the electron-bombardment engine in which mercury is used. A third type, the colloidal-particle engine, is less advanced. Although the first two engines may be considered to have had their potentialities well demonstrated, future work is required to

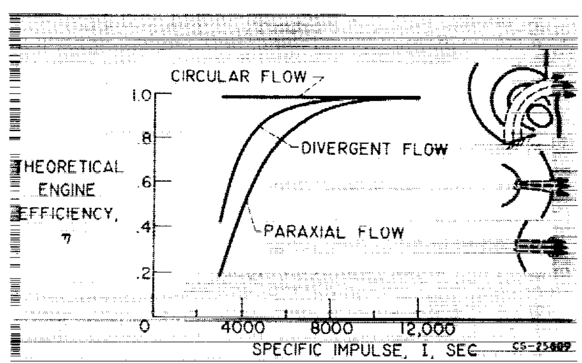

FIGURE 48(c)-3.—Theoretical efficiency of contact-ionization engines.

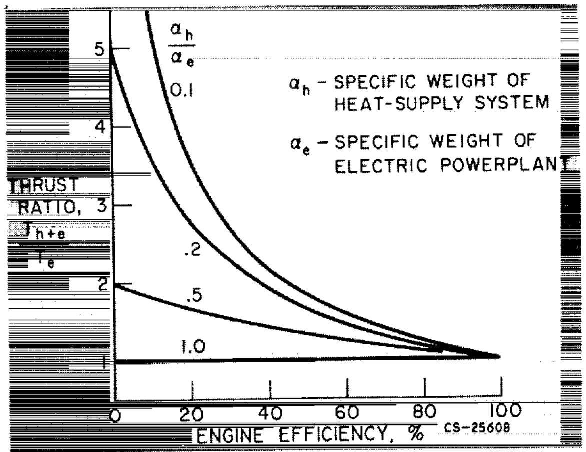

FIGURE 48(c)-4.—Ion-engine thrust gains with direct heat supply. Constant total weight and specific impulse.

improve the factors of thrust-to-weight ratio, efficiency, and durability.

In the contact ion engine, the principal source of energy loss is the heat radiated from the hot ion emitter. To some extent the accelerating electrodes can serve as radiation shielding to reduce this loss. Figure 48(c)-3 (from ref. 2) shows how greatly this might increase efficiency if the only radiation loss is assumed to be directly from the emitter to empty space; radiation falling on the accelerator is assumed to be perfectly reflected to the emitter. Compared with the "standard" paraxial-flow engine, the divergent-flow engine with its increased emitter current density is a substantial improvement. In a curved-beam configuration, the emitter radiation has no straight-line path of escape, and the efficiency calculated according to these assumptions approaches 100 percent.

Inasmuch as the principal source of loss is from the heat radiated by the emitter, the next logical question is whether there might not be a more effective (i.e., lighter) system for providing this heat. If, for example, the powerplant incorporates a very-high-temperature nuclear reactor, it might easily be possible to transfer the heat directly rather than first converting it to electric power. A separate nuclear reactor that would serve only to heat the emitter might even be considered. In figure 48(c)-4 is shown the effect of using a separate heat supply. The value 0.2 might be about right for the ratio of specific weights of the heat supply to the electric powerplant. As should be expected, the gains in engine output at a fixed specific impulse and total powerplant weight become greater at the lower engine efficiencies.

The electron-bombardment engine of the type invented and developed by Mr. Harold R. Kaufman at the Lewis Research Center is shown as a cutaway version in figure 48(c)-5. Mercury vapor is admitted to the ion chamber at a controlled rate. The hot cathode in the center emits electrons, which are attracted to the peripheral anode. An axial magnetic field extends the electron paths increasing the probability of ionizing collision. In operation the chamber is filled with a dilute plasma from which ions are extracted by a high potential difference impressed across the pair of perforated plates labeled "screen" and "accelerator." The principal sources of power loss include the heating power to the filament, the "discharge" power between filament and anode, and the power to maintain the magnetic field. Work under way is aimed at reducing each of these.

The electron-bombardment engine differs in one very important respect from the contact-ion engine. The propellant utilization—the fraction of the total propellant flow that becomes ionized and accelerated into the beam—is substantially less than 100 percent. Correct-

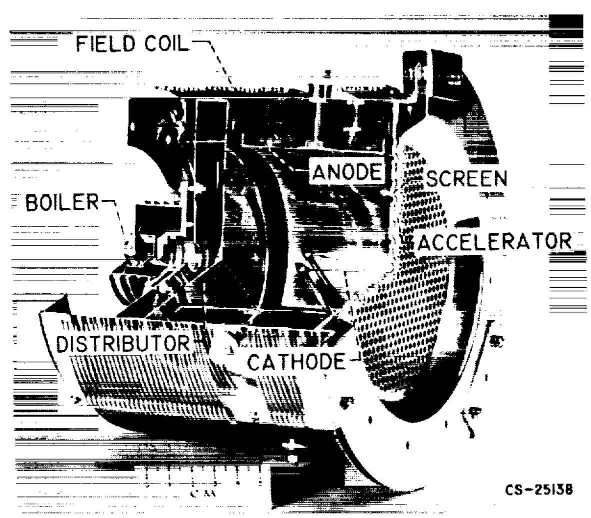

FIGURE 48(c)-5.—Electron-bombardment ion engine.

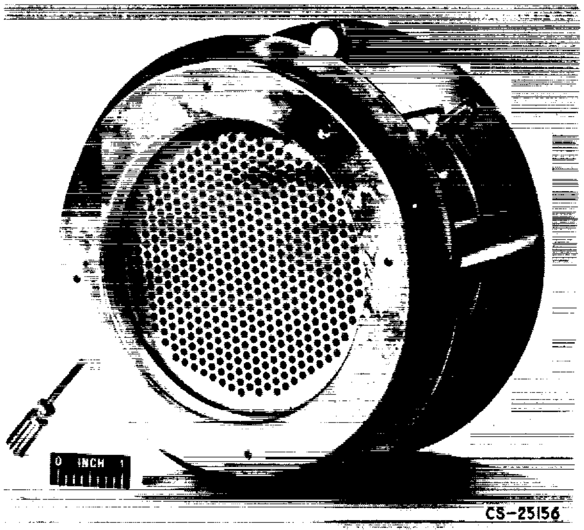

FIGURE 48(c)-6.—Permanent-magnet engine.

ing the efficiency and the specific impulse for this factor is easy enough; the problem arises from the neutral atoms going through the accelerator structure. A number of these neutrals will undergo charge exchange with the fast ions and fall into the accelerator. Ensuing erosion of the accelerator will then impose a durability limit on the entire system. Data have been obtained (ref. 8) which indicate that this charge-exchange process is indeed the primary contributor to the accelerator impingement and that the erosion is proportional thereto. It then follows that the life expectancy of a particular engine can be fairly well calculated in advance and that long life may require the engine to operate inefficiently or at a low thrust-to-weight ratio. Kaufman (ref. 8) shows that lifetimes of approximately 1 year (10,000 hr) may be predicted from current accelerator types with a thrust-to-weight ratio for the thrustor alone of about 0.001. A more severe problem may be the durability of the cathode. To date the main effort has involved the use of elemental (tantalum or tungsten) cathodes. Work now in progress with low-work-function cathodes should reduce power requirements and also promise extended cathode life.

In figure 48(c)-6 is shown an engine with permanent magnets in place of the field coil. The magnets are in contact with sheets of ferromagnetic material at the front and the back of the engine; thus the requisite weak field is induced throughout the ionization chamber. Not only does this procedure eliminate the power requirement of the field coil, but also, because it reduces by one the number of separate power supplies, simplifies and lightens the power-conditioning equipment. The weight of this engine corresponds closely to the weight of the equivalent configuration with field coil.

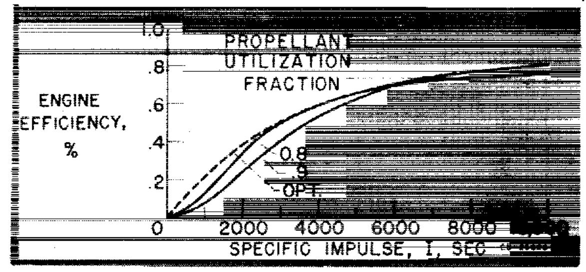

FIGURE 40(c)-7.—Engine efficiency.

The efficiencies obtainable from an electron-bombardment engine according to Kaufman's estimates are shown in figure 48(c)-7. These estimates were obtained by a realistic combination of the ingredients which have been separately determined. Permanent magnets were assumed, as were low-temperature cathodes. The optimum propellant utilization varies with specific impulse; the estimated efficiency attains only about 80 percent even at 10,000 seconds. By some standards, the efficiencies at the lower specific impulses are high. Nevertheless there seems to be room for improvement.

The efficiency estimates might be compared with some others, as in figure 48(c)-8. Here the curve for the bombardment engine and the curve calculated for the divergent-flow engine

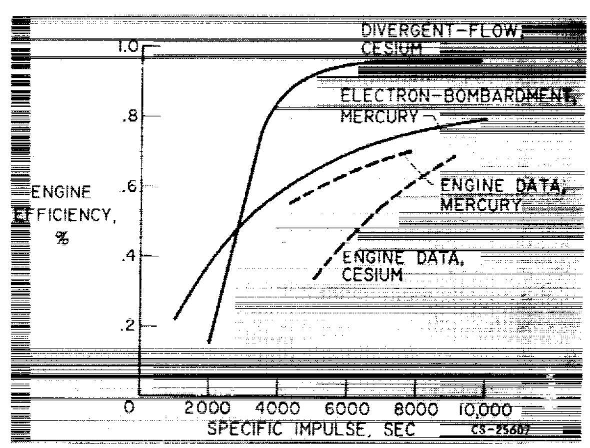

FIGURE 48(c)-8.—Estimates of ion-engine efficiencies.

are compared with some actual data for existing engines. The mercury engine data approach quite closely the estimates of attainable performance. The cesium engine data fall far short of the predicted divergent-flow engine performance. If the predictions are soundly based, some very substantial improvements in the low specific impulse range may be obtained. It should be pointed out that the divergent-flow curve is entirely theoretical; no engine data for such a configuration are available at this time.

The third category of electrostatic thrustor offers distinct promise for high efficiency in the low-specific-impulse region. Although its development is in an embryonic state, the colloid rocket, which uses particles of 100,000 atomic mass units or more may perform as well at low specific impulse as does the electron-bombardment engine at high specific impulse. The possibilities of this propulsion system have been discussed frequently during the past 5 years. Difficulties have been encountered in the production, charging, and acceleration of such tiny particles. A typical particle diameter might be about 40 angstroms. One frequent problem is the production of atomic or molecular ions, which, at the high accelerating voltages, consume much more power than their thrust is worth. Progress has been reported by Norgren (ref. 9). His process is the condensation of low-density vapor in an expanding nozzle. Figure 48(c)–9 shows both an electron photomicrograph of a sample of such colloids and also the particle-size distribution measured therefrom. The peak of the distribution occurs at about 0.009 micron. The particles were charged and accelerated, although only to 30 kilovolts, which gives an effective specific impulse of about 420 seconds. The progress with the colloid rocket is promising both because of the ability to provide high efficiency at low specific impulse and of a possible application with an advanced powerplant that gives megavolt potentials directly. This application is discussed by Mr. Edmund E. Callaghan.

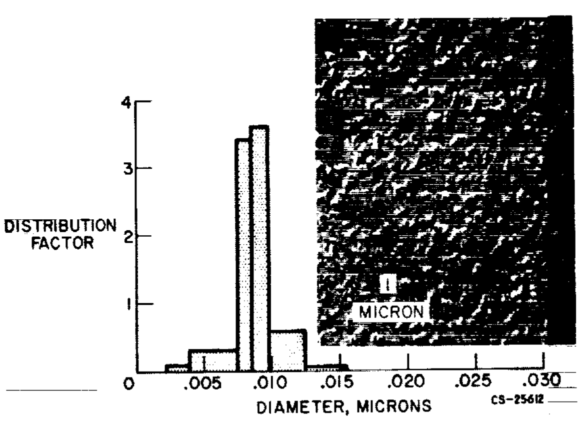

FIGURE 48(c)–9.—Particle distribution of mercurous chloride.

FUTURE CONTINGENCIES

The existing thrustors of both the cesium-tungsten and the electron-bombardment engine types will, of course, continue to be improved. The colloid rocket will be explored until its potentialities are more rigorously defined. Aside from such straight-line extrapolations, what may the future hold for electric propulsion? A great deal of effort is currently being devoted to powerplant development. The interface area between powerplant and thrustor, which contains the power conditioning and control equipment, has as yet received little concentrated attention. Such equipment will be subject to most stringent requirements as to weight and reliability. Further, the type of powerplant output may well be different from that which is now being designed. System optimization may demand that the characteristics of the thrustor be compromised in the interest of compatibility with other components. The possibility of a powerplant's yielding megavolts directly has certainly stimulated interest in the colloid engine wherein such voltages might be directly employed. A high-temperature nuclear turbogenerator system could similarly stimulate research in the cesium-tungsten engine with the heat-transfer system for the emitter.

Despite the vast number of missions that have been calculated, it should not be assumed that the field is exhausted. Special mission requirements may bring about the need for "nonconventional" electric thrustors. Already, as Mr. W. E. Moeckel has shown, the Jupiter mission might require a duration (and durability for the thrustor) of about twice the current target of 10,000 hours. Such durability requirements may demand new approaches.

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49. Advanced Concepts

By Edmund E. Callaghan

EDMUND E. CALLAGHAN, *Assistant Chief of the Electromagnetic Propulsion Division at the NASA Lewis Research Center, has specialized in research on icing problems, heat transfer, noise, turbulence, and cryogenics and currently is concerned with magnetics. In December 1961 he received the NASA Superior Achievement Award for outstanding accomplishment in his field. Mr. Callaghan attended Rensselaer Polytechnic Institute, where he received his B.S. degree in Aeronautical Engineering in 1943. He is a member of the Institute of the Aerospace Sciences and the American Physical Society.*

The need for ever increased payloads and decreased trip times has been emphasized by previous papers. Experience indicates that each major step forward requires development of some new power source. A prime example of this is the gas turbine, which has, to a great extent displaced the piston engine in air travel. Therefore, "advanced concepts" really means the study and development of energy sources that can be adapted to propulsion devices.

First, a propulsion concept, proposed in reference 1, which is based on a radioisotope electrogenerator integrated with a colloidal particle electrostatic engine will be considered briefly. Emphasis will be placed primarily on the electrogenerator since the colloidal particle electrostatic engine has been discussed previously by Mr. Warren Rayle. Secondly, the concept of a thermonuclear rocket will be explored. Since a wide variety of scientific and space uses of intense magnetic fields is anticipated, NASA research program on magnetics (ref. 2), low-temperature physics, superconductivity, and other related programs will be discussed in some detail.

Figure 49–1 illustrates the general mode of operation of a direct nuclear electrogenerator cell. Charged particles are emitted by a decaying radioisotope. Some of these particles will reach the collector, and the potential or charge

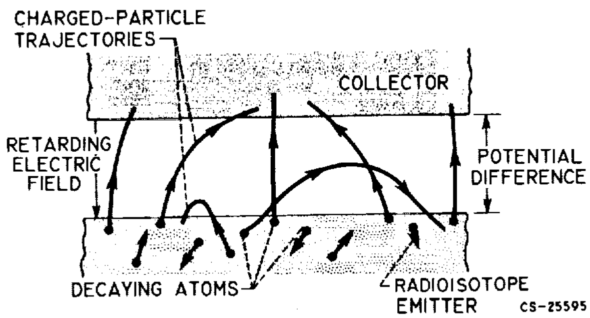

FIGURE 49–1.—Radioisotope electrogenerator cell.

will gradually build up until there is a sufficiently large potential difference between the collector and the emitter to prevent any further particles from reaching the collector. In this manner, a cell or voltage generating device is formed. In actual use an external load would be connected between the collector and emitter to complete the circuit for the flow of charge.

Not all the nuclear energy is converted to electricity. The random direction of emission of the particles causes some to have an insufficient velocity component oriented parallel to the field; thus, these particles do not reach the collector and fall back into the emitter. Since particles lose energy on passage through matter and the fuel layer cannot be infinitesimally thin and still produce power, particles emitted below the fuel layer surface lose energy

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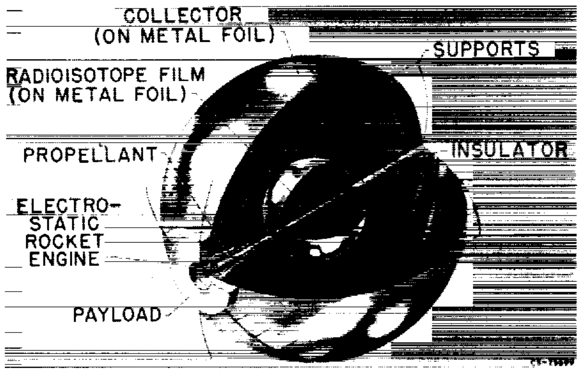

FIGURE 49–2.—Design study of 500-kilowatt electrostatic propulsion system with direct nuclear electrogenerator.

in the fuel layer. In addition, any excess energy of the particle when it reaches the collector is dissipated on penetration into the surface. In all cases where energy is lost in the emitter or collector, it is transformed into heat; this results in reduced efficiency and a heat disposal problem.

Design studies have been made of several conceptual spacecraft (ref. 1) using the ideas outlined herein and in Mr. Rayle's paper. One of these concepts is shown in figure 49–2. A spherical configuration has been assumed with an alpha-emitting radioisotope film located on the surface of the inner sphere; the outer sphere acts as the collector. In this concept the outer collector is floated at space electric potential, and the inner sphere then carries a negative charge. In order to evaluate this particular concept, i.e., to estimate its specific weight, it was decided to study an electrogenerator of 500-kilowatt capability. The radioisotope chosen was polonium 210, although cerium 144 would probably serve equally well. The collector for this case is about 135 feet in diameter, and the inner emitter diameter is about 45 feet. The radioisotope total mass required is about 30 pounds. Detailed design studies show a total system weight for the integrated electrogenerator and propulsion system of about 265 pounds or a specific powerplant weight of about ½ pound per kilowatt.

Recent studies have shown that the size of the system can be easily scaled downward to power levels of the order of 30 kilowatts without any increase in specific system weight. This makes the system of considerable interest for interplanetary space probes. The use of cerium instead of polonium would require radiation shielding of the payload package. For a typical configuration, this represents an additional weight of perhaps 1200 pounds, which would give a specific system weight of 3 pounds per kilowatt. Since this represents an order-of-magnitude reduction in specific weight over current systems and, hence, greatly enhanced emission capabilities, it becomes important to study the physics and engineering problems associated with the development of such a spacecraft.

Basic physics problems exist in regard to (a) whether particles will knock electrons out of the support foil and cause local shorts, (b) the flux distribution of particles coming out of the support foil, (c) the sputtering of the collector and emitter support materials, (d) and high voltage breakdown between the collector and emitter. Some of the engineering problems to be solved are the packaging of the isotope during launch and the dissipation of its high heat output. The mere obtaining of large quantities of radioisotopes other than cerium may be a serious problem. There are the problems of integrity of the package if the mission is aborted, the production of lightweight insulators with small leakage, and the development of methods for depositing and attaching isotopes in thin films to the emitter surface.

Study has been started on some of these problems, but as yet the problems have not been investigated thoroughly. Several small experiments to investigate some of the basic physics problems are under way.

The next advanced concept to be considered is the fusion engine. Considerable effort has already been expended on the development of fusion power. This program, known as Project Sherwood, is being conducted by the Atomic Energy Commission with the express aim of obtaining a controlled thermonuclear reaction. In principle, what must be done is understood, but the problems are difficult. When it is considered that fusion is similar to building and maintaining a small sun, the difficulties are quite apparent.

The principle of achieving thermonuclear fusion reactions is well known; all that is necessary is to cause the atomic nuclei of light elements to collide with sufficient energy. The problem is that "sufficient energy" means that high enough relative velocities must be achieved to overcome the electrostatic repulsion that tends to keep the nuclei apart, and to achieve these velocities a plasma must be created with temperatures of the order of tens or hundreds of millions of degrees Kelvin. At these temperatures the nuclei move about randomly with high enough velocities to combine and release sufficient amounts of energy to enable the reaction to be self-sustaining.

It is clear then that the two major problems are (1) to heat a plasma of light elements to a sufficiently high temperature to cause fusion and (2) to contain the plasma without the use of material walls. The AEC work on both plasma containment by magnetic bottles and plasma heating by a wide variety of methods has not been specifically aimed at a space propulsion system. With this in mind, therefore, some typical fusion reactions will be considered.

As shown by the following equations, two deuterium nuclei fuse together to produce either helium 3 and a neutron or tritium and a proton. These two reactions have an equal probability of occurrence. A deuterium nucleus and a tritium nucleus combine to produce helium 4 and a neutron, and a deuterium nucleus combines with helium 3 to produce helium 4 and a proton. Of interest here is the extremely high energy of the reaction products.

$$D+D \rightarrow He^3 \text{ (0.8 Mev)} + \text{neutron (2.4 Mev)}$$
$$D+D \rightarrow T \text{ (1.0 Mev)} + \text{proton (3.0 Mev)}$$
$$D+T \rightarrow He^4 \text{ (3.4 Mev)} + \text{neutron (14.1 Mev)}$$
$$D+He^3 \rightarrow He^4 \text{ (3.6 Mev)} + \text{proton (14.7 Mev)}$$

The major task, therefore, is to examine these reactions and to determine how they can best be utilized in space. It is evident, for example, that the last two reactions release far more energy than either of the deuterium reactions (about 18 Mev compared with 3 or 4 Mev). Consider further that the deuterium-tritium reaction produces a helium 4 nucleus and a neutron, whereas the deuterium-helium 3 reaction produces a helium 4 nucleus and a proton, both of which are positively charged. If the use of deuterium-tritium is studied as an energy source, most of the available energy is in the neutrons that escape the magnetic bottle. The high-energy neutrons must be trapped in thick and massive shields and their energy removed as heat. This brings about the usual thermodynamic problems associated with the utilization of a heat source in space. Either this heat must be used directly to heat a propellant or it must be converted to electricity and thence into an electrical propulsion system. Studies reported in reference 3 show no substantial gains as compared with an ordinary fission reactor. Therefore, the $D-He^3$ reaction is left, the products of which are charged particles and, hence, can be contained magnetically. If for the moment the problem of plasma heating is ignored and the necessary conditions are assumed to be created, the problem of magnetic containment must be considered. Early studies of thermonuclear propulsion showed that ordinary electromagnets, for example, of copper operating at room temperature, were far too massive for any practical use in space. Later studies in which cryogenically cooled magnets were used to decrease resistivity resulted in substantial gains, but such systems did not show any marked improvements over other space power and propulsion systems that were already being developed. In the last several years tremendous advances have been made with superconducting materials and superconducting magnets. In fact, in the laboratory, fields of the order of 90 kilogauss have been achieved with superconducting magnets. Even more spectacular gains will probably be made in the future. In any event, the use of superconducting magnets for plasma containment is a distinct possibility, and the conceptual design of a thermonuclear rocket is based on their use.

It should be emphasized that it is not as yet known how to build a thermonuclear reactor but that enough is known about the kinds of energy releases to analyze the possible propulsion implications for advanced space missions.

The basic elements of a propulsion system using $D-He^3$ as a fuel are shown in figure 49–3. This figure illustrates one of the many concepts explored at the Lewis Research Center (ref. 3).

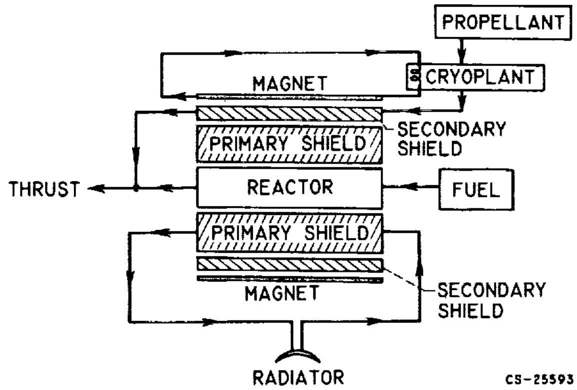

FIGURE 49-3.—Final two-shield system for a reaction.

The view shown is a cross section of a cylindrical system using a solenoidal magnet. For a device such as envisioned here, magnetic fields of the order of 100,000 gauss in the center and 200,000 gauss at the ends are required to contain the plasma and to keep the overall size to reasonable values.

For an assumed reactor whose total power output is 100 megawatts, the total size of the reactor is found to be only 50 centimeters in diameter and 250 centimeters long for the conditions chosen.

Note in the figure that several shields have been placed between the reactor and the magnet. These shields are necessary since a $D-He^3$ plasma radiates gamma rays, which, if allowed to impinge on the superconducting magnet, would cause excessive heat loads that must be removed at the extremely low temperature (4° K) at which the magnet operates. For the particular device that will be discussed, liquid hydrogen is used as a propellant that is heated in the secondary shield and mixed with the reaction products and expelled to product thrust. The use of a propellant separate from the fuel is necessary in order to produce the thrusts and specific impulses that are more nearly optimum for interplanetary flights. It should be pointed out that very little is known about the mixing and expulsion of gases at the temperatures involved. This is an area that is currently of great interest in plasma physics.

Although the superconducting magnet will not generate any heat by internal resistance, some heat will be generated by the residual gamma rays that pass through both the primary and secondary shields. This heat must be removed by the cryoplant. For the conceptual design it has been assumed that the magnet is cooled by liquid helium and that the heat from gamma-ray absorption in the magnet is handled by the cryoplant, which rejects its heat to the liquid-hydrogen propellant. The heat generated in the primary shield is rejected to space directly. It could, of course, be used to generate auxiliary power for nonpropulsive applications aboard the spacecraft.

Any analysis of a system such as this is dependent on a large number of variables. For the configuration shown and in the range of specific impulses of interest, specific powerplant weights considerably less than 1 pound per kilowatt and thrust-to-weight ratios of the order of 0.004 pound of thrust per pound of propulsion system weight appear to be achievable (ref. 3). Since these values, like those for the isotope cell system, are at least an order of magnitude better than electric propulsion systems currently under development, it is clear that thermonuclear propulsion offers exciting possibilities for the conquest of space if and when the reaction is achieved.

The question arises then as to what is being done to solve the many problems that are involved. The two principal stumbling blocks are plasma heating to thermonuclear temperatures and containment. As mentioned before, the AEC has a large effort covering many phases of controlled thermonuclear reactions. At the Lewis Research Center, work is being done on magnetics as related to containment (ref. 2) and on plasma heating, as well as on a wide area of plasma physics. Currently a large experiment is being set up that uses a heating method called "ion cyclotron resonance," which was first analyzed in reference 4. Radio frequencies are used to excite ions to very high temperatures. Several preliminary experiments using this technique are under way, and the results to date are quite promising.

Although strong magnetic fields are essential for the fusion process, their applications to other areas may be even more fruitful. Obvious applications of such fields are in magnetohydrodynamic power generation, magnetic nozzles, and the magnetic shielding of spacecraft from high-energy charged particles. Because of these interests, the remainder of this paper will be devoted to discussing the Lewis program in magnetics.

Magnetohydrodynamic power generation is a particularly interesting concept since it avoids some of the problems associated with rotating power equipment at extremely high temperatures. It does, however, have unique problems of its own and appears to be feasible for space only if the required magnetic field can be generated by using superconductors.

The concept of shielding interplanetary spacecraft from high-energy particles by using superconducting magnets has been explored by several investigators. It appears that for large interplanetary vehicles an order-of-magnitude weight savings may be possible as compared with the usual mass shielding.

Perhaps even more important are the wholly scientific aspects of high fields. For example, studies are being made of the biologic effects of extremely high fields, particularly as related to cancer.

The achievement of intense magnetic fields for steady-state operation and over reasonably large volumes is in itself a considerable accomplishment. Space applications add the additional requirements of light weight and low power consumption.

It has become increasingly evident that minimum power, both for the magnet and indeed for the total system, can be achieved only by means of cryogenics. One of the goals is to construct and operate large cryogenically cooled magnets made of both normal and superconducting materials and to apply these magnets to the applications previously mentioned.

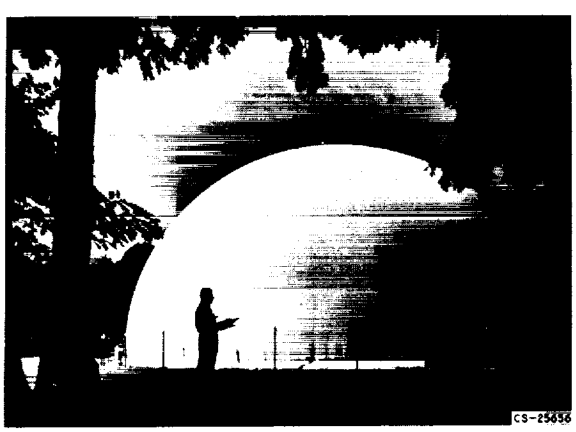

FIGURE 49-4.—Helium storage bag.

This kind of an activity requires the availability of an electric power source for magnet coils and large quantities of cryogenic fluids. The power source is a homopolar machine. This is an interesting device that uses liquid brushes of a sodium-potassium mixture and is capable of producing upwards of 200,000 amperes at 15 volts for 1 minute. It can be operated at 2.2 megawatts or less continuously and is currently being used to power two water-cooled magnets with fields up to 110,000 gauss.

For cryogenic fluids, a helium liquefier of 100-liter-per-hour capacity is available, and a liquid-neon system with 60-liter-per-hour capability will soon be ready. It is necessary, of course, that both of these gases be recaptured after they have served as coolants and have been vaporized. For example, a large plastic bag is used for helium storage, as shown in figure 49-4. This bag is an inflatable structure; the outer shell is held up by air pressure, and the inner bag holds the helium. This kind of system can achieve greatly reduced costs over metal tanks.

In order to develop either superconducting or cryogenic magnets, the characteristics of applicable materials in the presence of both low temperatures and high magnetic fields must be studied. Figure 49-5 shows a 100-kilogauss water-cooled magnet into which a helium Dewar has been lowered. Provisions have been made to pump on the Dewar, which reduces the helium vapor pressure and, hence, achieves temperatures down to about 1° K. A working volume in the Dewar of about ¾ inch in diameter by 12 inches high is available. A larger system, which provides fields over 100 kilogauss with a working volume of 3 inches in diameter by 12 inches high, is also being used. Both these facilities are used for studies of effects of strong fields and low temperatures. Of particular interest, however, are the superconducting properties of such materials as niobium-tin and niobium-zirconium alloys. These have been studied, and a number of small superconducting magnets have been constructed.

FIGURE 49–5.—Magnetoresistance and superconductivity test apparatus.

In the area of superconductivity, a wide variety of research problems requires solution. One of the most interesting of these problems is the "training effect." A superconducting magnet of niobium-zirconium alloy will "learn" to build higher and higher fields. Each time it is operated, the wires will carry successively higher currents and, hence, produce higher fields until a maximum is reached. Another effect, which is not yet understandable, is the fact that the maximum current a given superconducting wire will carry decreases with increasing length.

At the current state of the art many metallurgical problems exist. Material properties, supposedly the same, will vary from sample to sample. It is quite difficult to achieve reproducible results with niobium-tin magnets.

If the construction of cryogenic but nonsuperconducting magnets is considered, it is well known that the electrical resistance of materials decreases with decreasing temperature. At low temperatures, however, two effects are encountered that are normally negligible at room temperature. When the temperature is reduced sufficiently and the lattice structure of the material becomes relatively stable, the regularity of the lattice becomes important. Regularity of the lattice can be achieved by using very-high-purity materials. The second effect is called magnetoresistance and is due to the magnetic field in which the conductor lies. If both low temperatures and very pure materials are used, the magnetoresistance effect dominates. For the construction of cryogenic coils it is desirable to choose a material that has a minimum magnetoresistance in the presence of high fields. Studies have shown that either very pure aluminum or sodium gives the minimum total resistance at low temperatures and in high fields.

Based on this information and on a series of heat-transfer studies, a large magnet is being constructed that will provide fields up to 200 kilogauss for several minutes (ref. 5). Several single experimental coils have been constructed; one is shown in figure 49–6. This coil has a bore of 12 inches and an outer diameter of over 3 feet. The coil is made of a spiral wrap of very high purity aluminum (99.9983 percent) laminated to stainless steel. Small stainless-steel separators are used to transfer the forces and to provide cooling passageways. The size of the outer restraining hoop indicates the kinds of stresses anticipated; 200-kilogauss fields will give stresses equivalent to an internal gas pressure of about 25,000 pounds per square inch.

FIGURE 49–6.—Liquid-neon-cooled coil.

Cooling is accomplished by nucleate boiling of the cryogenic fluid in which the coil is placed. Tests have been conducted in both nitrogen and hydrogen. In the final configuration a stack of 12 of these coils will be cooled by liquid neon to provide a field of 200 kilogauss. This magnet should be very useful for studies of various physical phenomena in intense fields. Further in the future a magnetic-mirror machine using 36 coils will be constructed. This machine should provide a central field of 100 kilogauss with end mirrors of 200 kilogauss and will be used in fundamental studies of plasma heating and containment.

To summarize briefly, two proposed conceptual propulsion systems that appear to give far greater mission capability than others that are under development have been reviewed. Both, however, have serious problems that must be solved before any real feasibility can be demonstrated. A wide variety of uses is anticipated for intense magnetic fields for future space, ground, and scientific applications. In space, however, it will probably be necessary to achieve these fields using superconducting systems. Experience at Lewis and other laboratories throughout the country is disclosing exciting new possibilites for superconductors.

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